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SOLAR POWER SATELLITE

SYSTEM DEFINITION STUDY

Space Transportation Analysis

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16. Abstract

This document contains reports on the analysis of the "small" HLLV (120 MT payload vs. 400 MT for the "large" HLLV, the shuttle-derived transportation system concept, and the electric orbit transfer vehicle (EOTV).

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FOREWORD

The SPS System Definition Study was initiated in June of 1978. Phase I of this effort was completed in December of 1978 and was reported in seven volumes (Boeing document number D180-25037-1 through -7). Phase II of this study was completed in December of 1979 and was completed in five volumes (Boeing document number D180-25461-1 through -5). The Phase III of this study was initiated in January of 1980 and is concluded with this set of study results published in five volumes (Boeing document number D180-25969-1 through -5):

- Volume 1 - Executive Summary
- Volume 2 - Final Briefing
- Volume 3 - Laser SPS Analysis
- Volume 4 - Solid State SPS Analysis
- Volume 5 - Space Transportation Analysis

These studies are a part of an overall SPS evaluation effort sponsored by the U. S. Department of Energy (DOE) and the National Aeronautics and Space Administration (NASA).

This series of contractual studies were performed by the Large Space Systems Group of the Boeing Aerospace Company (Gordon Woodcock, Study Manager). The study was managed by the Lynden B. Johnson Space Center. The Contracting Officer is David Bruce. The Contracting Officer's Representative and the study technical manager is Tony Redding.

The subcontractors on this study were the Grumman Aerospace Company (Ron McCaffrey, Study Manager) and Math Sciences Northwest (Dr. Robert Taussig, Study Manager).

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ABBREVIATIONS AND ACRONYMS

db	Decibels
EOTV	Electric Orbit Transfer Vehicle
ET	External Tank
Ft	Foot
GEO	Geosynchronous Earth Orbit
GLOW	Gross Liftoff Weight
HLLV	Heavy Lift Launch Vehicle
Isp	Specific Impulse
JSC	Johnson Space Center
K	Kilo
kg	Kilogram
L/D	Length/Diameter ratio
LEO	Low Earth Orbit
LH_2	Liquid Hydrogen
LO_2	Liquid Oxygen
M	Meters
MPD	Magneto Plasma Dynamic
MT	Metric Ton = 1000 Kilograms
M/S	Meters per Second
MSFC	Marshall Space Flight Center
$M	Millions of dollars
NM	Nautical Miles = 6076 ft.
POTV	Personnel Orbit Transfer Vehicle
psf	Pounds per Square Foot
SOC	Space Operations Center
SPS	Solar Power Satellite or Space Power System
SSME	Space Shuttle Main Engine
SI	Standard Internationale
TFU	Theoretical First Unit cost

TRANSPORTATION SYSTEMS ANALYSES

1.0 INTRODUCTION AND SUMMARY

This report describes an investigation of alternative transportation options for the solar power satellite. The options include alternative Earth-to-Orbit transportation and further examination of electric orbit-to-orbit systems. Where the influences on the SPS and the transportation costs are discussed, the DOE/NASA silicon reference SPS (Reference 7) has been assumed.

1.1 PROBLEM STATEMENT

The earliest studies of large launch vehicles were conducted in the mid-1960's during the development of Saturn V. With the initiation of shuttle development, such studies were for a time dropped. As concept development for the solar power satellite began, there again developed an interest in large launch vehicles. Boeing developed a concept of a 500,000 lb. payload single stage-to-orbit ballistic vehicle in 1974. It used dual-fuel propulsion with oxygen-hydrocarbon and oxygen-hydrogen engines. A later study, funded by NASA-JSC and MSFC, examined heavy lift launch vehicles and concluded that staged ballistic configurations would have a cost advantage over single staged systems. At that time SPS payloads were thought to have very low density, on the order of 20 kilograms per cubic meter. Consequently, the configurations of that time period employed very large expendable shrouds.

Development of space fabrication concepts improved the payload density to about 75 kilograms per cubic meter and the launch vehicles were resized in response. JSC, in 1977, developed a winged vehicle concept for horizontal land landing. A comparative assessment of this versus the sea-landing ballistic system showed that the land lander would be operationally preferable and about equal to cost to the ballistic system, but that the specific configuration had inadequate payload volume. It was subsequently reconfigured to increase payload volume and became the reference system. The evolution discussed here is shown in Figure 1.1-1.

During all of this, the question of the "right" vehicle for SPS, especially the "right size," was never specifically raised.* The aims of the studies were to evaluate the performance and cost potentials of large vehicles and to compare winged runway landers with ballistic sea landers. (Winged vehicles were selected for their better operational characteristics, i.e., shorter turnaround time.)

The reference SPS HLLV has an estimated payload capability of 420 metric tons and a liftoff mass of 11,000 metric tons. It is between 3 and 4 times as massive as the Saturn V moon rocket and nearly six times as massive as the Space Shuttle. Its large size and development cost have become an SPS cost issue. Further, it is too large to be on an evolutionary path from the Shuttle. (It does use the SSME in the second stage.)

*An early parametric study by Dan Gregory of Boeing illustrated that an economically optimal size exists and suggested a range of 200 to 500 metric tons payload for the (then) SPS scenarios of 20,000 megawatts per year or more (the present DOE scenario is 10,000 megawatts per year).

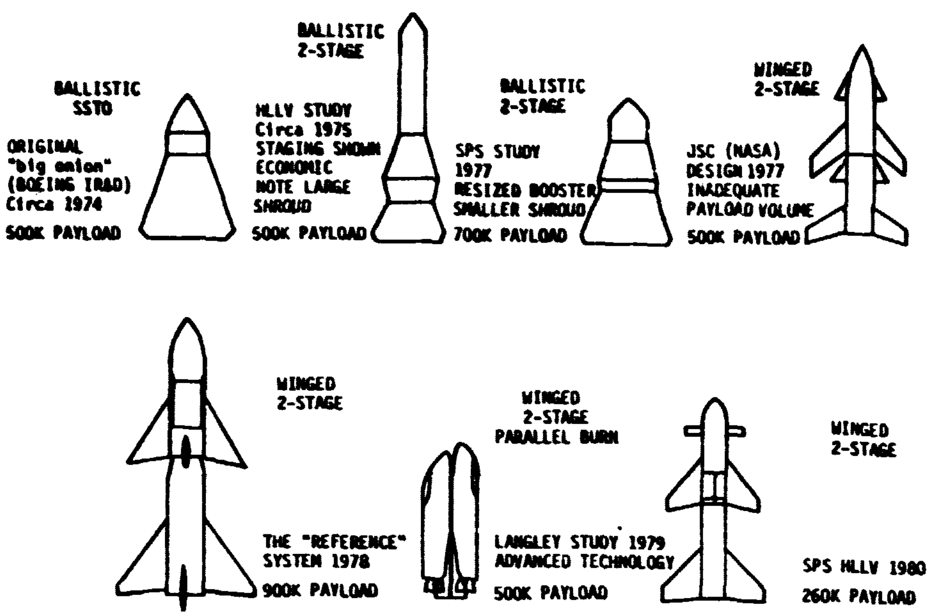

Figure 1.1-1 SPS Launch Vehicle Concept Evolution

The utility of smaller vehicles is an important question for the SPS evaluation studies now nearing completion. Accordingly, this study evaluated a "small" HLLV. Issues examined included performance, sizing, influence on SPS hardware packaging and construction operations, commonality with Shuttle subsystems and nonrecurring and recurring cost.

1.2 SUMMARY OF OPTIONS

There is, of course, no limit to the number of configurations and size options for launch vehicles. Figure 1.2-1 illustrates some of the winged and ballistic evolutionary paths that have been conceived. (The winged HLLV at the lower right is the reference vehicle). A range of sizes, payload volume and mass capabilities, and degrees of reusability are shown. This figure was originally prepared about two years ago to illustrate evolution potentials. At that time little work had been done on SPS development approaches and none of the alternatives were investigated in any depth.

The reference orbit-to-orbit system is an electric orbit transfer vehicle of roughly 300 megawatts power, 4000 tons delivery transfer payload, using argon as propellant for its ion engines. Recently, issues have been raised as to (1) thermal effects on array performance in low Earth orbit; (2) sensitivity of the system's cost and life to radiation degradation of the array and degree of annealing possible; (3) possible environmental effects arising from injection of argon ions into the Earth's magnetosphere. Accordingly, it was deemed desirable to perform a sensitivity analysis on the reference EOTV and to re-open the question of chemical (LO_2/LH_2) orbit transfer systems, especially options that might be derived from Shuttle hardware.

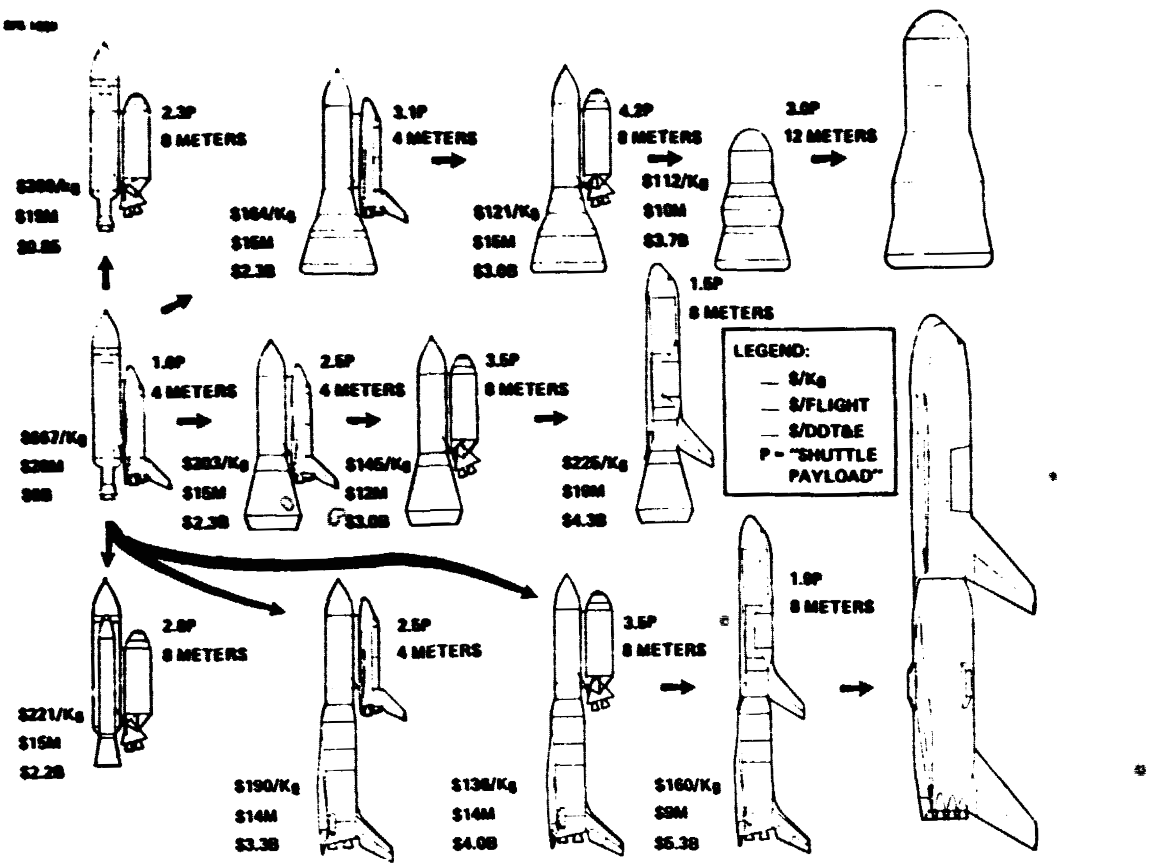

Figure 1.2-1. Potential Launch Vehicles

2.0 SMALL HEAVY LIFT LAUNCH VEHICLE

The present day use of the term "heavy lift" connotes a launch system with a payload capability substantially greater than the 30 tonnes of the Space Shuttle. A "small" heavy-lift system is a large vehicle; the term "small" is comparative to the very large SPS reference system.

2.1 SIZE AND CONFIGURATION SELECTION

A preliminary investigation was carried out to select tne appropriate size range and adopt a configuration approach.

2.1.1 Payload Volume and Mass Considerations

Certain of the hardware items in the reference SPS system were sized to take advantage of the large (17-m diameter by 23-m length) payload bay of the reference launch vehicle. Principal items are the electrical rotary joint (slip ring) and the crew habitats of the orbital bases. Clearly, a smaller payload bay volume will impose penalties on these elements of the system and require added construction labor in space. The realizable reduction of size of the launch vehicle without reduction of the large payload bay envelope would be extremely limited. Accordingly, it was necessary to make a reasonable judgment as to how much envelope reduction could be accommodated by SPS systems without excessive penalties. The electrical slip ring cannot be made appreciably smaller, given the existing requirements for currents, number of busses, and voltages. It is, however, a one-per-SPS unit and on-orbit assembly should not be an inordinate penalty with proper design. A smaller crew habitat will house fewer crew per unit, but there is nothing special about the 100-man reference capacity. Smaller habitats will incur operational inconveniences but will provide nonrecurring cost reductions and may avoid the necessity (presently shown in the reference SPS development scenario) to develop an intermediate-sized habitat (larger than SOC but smaller than the ultimate article) for a demonstration project.

Based on these and similar considerations it was concluded that the limiting article is the power transmitter subarray. There are more than 7000 of thse units for each SPS, they include most of the electronic complexity of the SPS (each subarray is fed by reference phase and data fiber-optic cables and by power supply cables), and they require high-precision mechanical assembly. The subarrays are 10.4 meters square by about 30 cm thick. Accordingly it was decided to employ a square-cross-section payload bay 11 meters square, with some convenient length. A study of technology requirements for Earth-to-GEO transportation system (performed by Boeing for Langley Research Center) developed configuration concepts for HLLV's in the 200-tonne payload range, control configured without central vertical tails (Reference 8). The configurations were quite amenable to aft-located,square-cross-section payload bays. It was decided to adopt this design approach.

The payload bay length was selected on the basis of performance and scaling considerations and density indications from previous SPS payload packaging studies. The effects of this smaller payload bay are discussed in detail in Section 2.3.

2.1.2 Performance and Scaling Considerations

The preliminary scaling analysis included consideration of the variation in structural efficiency with stage size and propellant load. Simplified analyses of vehicle performance are often based on the assumption of constant propellant mass fraction. This is a very

poor assumption for this class of vehicle. A better scaling rule is that the inert mass has a fixed and variable aspect. The variable part represents mass added as the propellant load is increased. The fixed part is constant for a given vehicle diameter but varies with diameter and other factors.

For this analysis, prior results were examined to select the "b" parameter (factor by which propellant load is multiplied to get variable inert mass); the "a" parameter was selected from the rough plot of a versus the square of diameter shown in Figure 2.1-1. (It is regarded as plausible that "a" is proportional to the square of diameter).

Based on the SPS reference vehicle and the smaller vehicles designed by the study for Langley, values of "a" were estimated as 140,000 kg and "b" as 0.08 for each stage. The "a" value corresponds to a 12-meter tank diameter. The stage inert mass is given by:

$$M_I = a + bM_p$$

where M_p is mainstage impulse propellant load. Second stage inert mass includes on-orbit maneuver propellant and booster inert mass includes post-separation and flyback propellant. Other assumptions are given in Table 2.1-1.

Initial sizing was based on a fixed ideal delta v to injection of 9200 m/sec (30,183 ft/sec). Given a fixed delta v, it is possible to represent the payload ratio for a parallel -burn vehicle without crossfeed as:

$$\frac{m_t}{P_1} = \frac{1-b_2(\mu_2-1)}{\mu_2(\mu_1-1)} - \frac{b_1[1-b_2(\mu_2-1)]}{\mu_2} + \frac{c_1}{c_2} r\left[\frac{1-b_2(\mu_2\mu_1-1)}{\mu_2(\mu_1-1)}\right] - \frac{a_2}{P_1} - \frac{a_1}{P_1}\frac{[1-b_2(\mu_2-1)]}{\mu_2}$$

where r is the ratio of orbiter to booster thrust, μ_1, and μ_2 are mass ratios of the parallel burn and orbiter alone burn respectively, p_1 is the booster propellant load, and c_1/c_2 is the ratio of booster to orbiter ISP.

The Isp of the parallel burn is given by:

$$\bar{c} = \frac{(1+r)c_1 c_2}{c_2 + c_1 r}$$

The mass ratios for each burn are computed from the Tsiolkovskii equation,

$$\mu = \exp\left(\frac{\Delta V}{I_{sp}}\right)$$

(In SI units the Isp is jet velocity in m/s. In conventional units Isp in seconds should be multiplied by g in the Tsiolkovskii equation).

For a series burn system, the payload is given by

$$m_L = \frac{1}{\mu_2}\left\{[1-(\mu_1-1)b_2]\right\}\left\{\frac{[1-(\mu_1-1)b_1]P_1}{\mu_1-1} - a_1\right\} - a_2$$

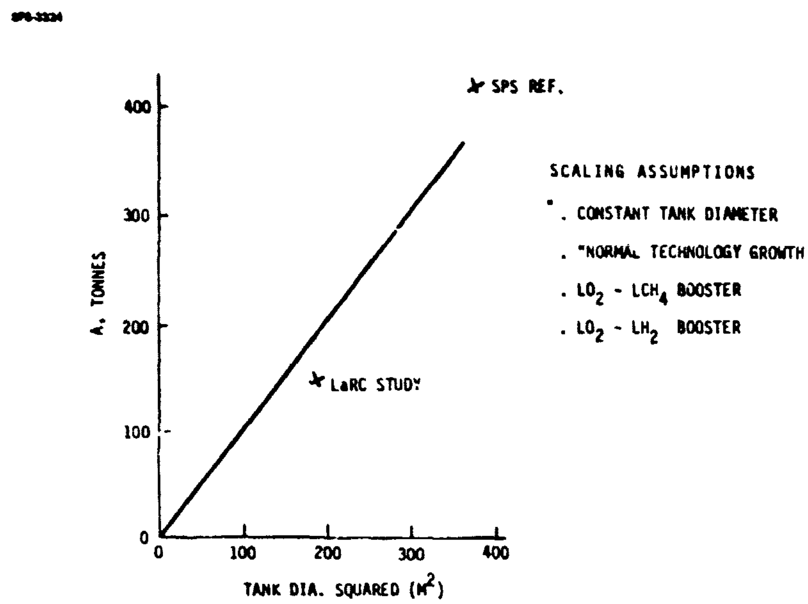

Figure 2.1-1 "A" Parameter Scaling

TABLE 2.1-1

SPS LAUNCH VEHICLE TECHNOLOGY ASSUMPTIONS

- LO_2-LCH_4 BOOSTER
- LO_2-LH_2 ORBITER
- ENGINE TECHNOLOGY CONSISTENT WITH SPACE SHUTTLE MAIN ENGINE SPECIFICATION
- CRYOGENIC ORBIT MANEUVERING PROPULSION
- IN SOME CASES, CONTROL-CONFIGURED AERODYNAMICS
- STATE-OF-THE-ART CONSTANT DIAMETER ALUMINUM TANKS; TITANIUM WHERE WARRANTED FOR AERO SURFACES: MODERATE USE OF COMPOSITES IN UNHEATED, DRY STRUCTURE
- SERVICEABLE SHUTTLE-TYPE THERMAL PROTECTION FOR ORBITERS
- REUSABLE LH_2 INSULATION
- SUBSYSTEMS GENERALLY CONSISTENT WITH SHUTTLE STATE-OF-ART
- EVOLUTIONARY IMPROVEMENTS IN SUBSYSTEMS SERVICEABILITY
- ONBOARD BIT/FIT

These equations were programmed on the minicomputer to plot payload and other pertinent parameters versus staging velocity for a range of total mass values. Results for series burn are shown in Figures 2.1-2 through 2.1-6. The parallel burn comparison for 4000 tonnes liftoff mass is shown in Figures 2.1-7 through 2.1-9.

The optima are relatively flat, i.e., insensitive. This results from the inert mass model. Use of a constant propellant mass fraction (λ') results in sharper optima. Cost optima will be at higher staging velocities than mass optima because (1) LH_2 is more expensive than hydrocarbon; (2) orbiters are more expensive than boosters.

In both instances, practical considerations require a staging velocity higher than the mass optimum. In the series burn case, it is necessary to have about twice the propellant load in the booster as the orbiter, or the booster becomes too short to arrive at a reasonable configuration (assuming booster tank diameter equals orbiter tank diameter). In the parallel burn case, the available thrust-to-mass ratio at staging forces a higher velocity. In both cases the minimum practical values is about 2750 m/s ideal, near 5000 ft/sec relative.

The ratio of payload mass to liftoff mass improves with larger vehicles (as one would expect). This is because the propellant fraction improves as propellant load is increased. Figure 2.1-10 shows the decrease in M_i/M_t as liftoff mass is increased. Points from the Langely study vehicles are also shown. The latter assumed parallel burn with crossfeed (from booster to orbiter) and would be expected to perform somewhat better than the vehicles represented here.

Based on these results, a liftoff mass of 4000 tonnes was selected for a point design study. The payload capability anticipted from these parametric analyses is 120 tonnes (series burn) or 100 tonnes (parallel burn). SPS packaging studies have indicated that the payload bay density (lift capability/volume) should be in the range 75 kg/M^3 to 100 kg/M^3. The forcing function is the relatively low density of transmitter subarrays; they average much less than 75 kg/M^3 but by mixing subarrays with high-density items, an average in the range stated is obtained. At 120 tonnes lift capability, an 11-meter-square payload bay cross-section requires a length of 13.2 m to reach 75 kg/m^3. Anticipating the 120 tonnes estimate to be slightly conservative, a length of 14m was selected. Note that this payload bay, although it has 5.6 times the volume of the shuttle payload bay, is actually about 4 meters shorter. Accordingly, a check was made to evaluate the propellant capacity of an orbit transfer vechicle constrained to these payload bay dimensions. Its propellant capacity was limited to about 230 tonnes see Figure 2.3-2). This was deemed adequate. (More volume-efficient OTV arrangements are possible).

The analysis conducted did not include booster flyback range as a parameter. For typical boosters, flyback propellant is 10% to 20% of inert mass; the variation of flyback propellant with staging conditions is a significant overall optimization parameter. Since staging velocity selection was downward -limited to 2750 m/s (ideal) by other factors, reducing staging velocity to reduce flyback range is not a consideration. Adjusting staging angle conditions to reduce flyback range remains an option. Flyback range may be approximated by the following algorithm:

Orbit semimajor axis: $$a = \frac{r}{2 - \frac{rv^2}{\mu}}$$

where v is inertial velocity, r is radius from Earth's center at staging, and μ is Earth's geopotential.

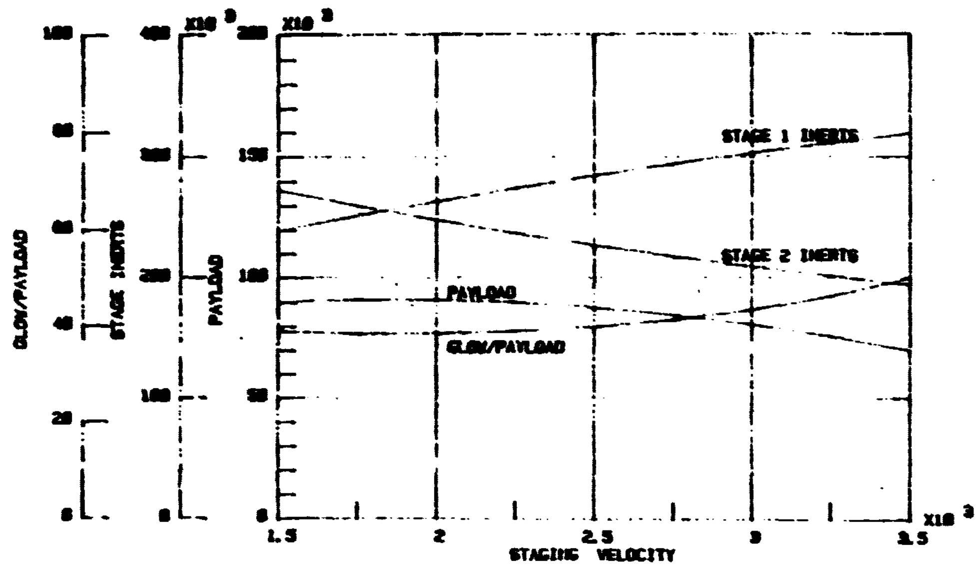

Figure 2.1-2. Series Burn–Glow = 3.5E6

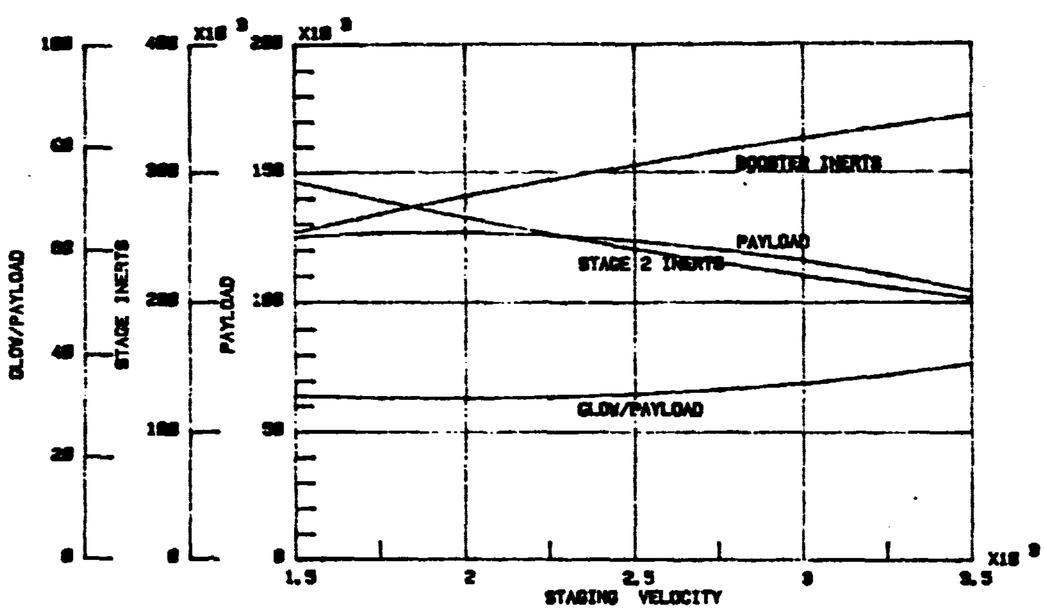

Figure 2.1-3. Series Burn–Glow = 4E6

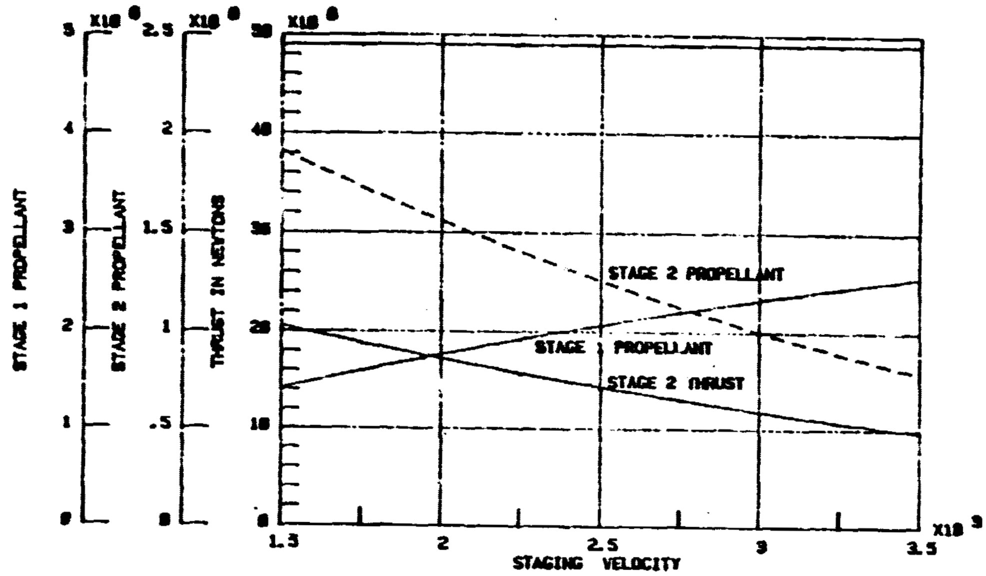

Figure 2.1-4. Series Burn–Glow = 4E6

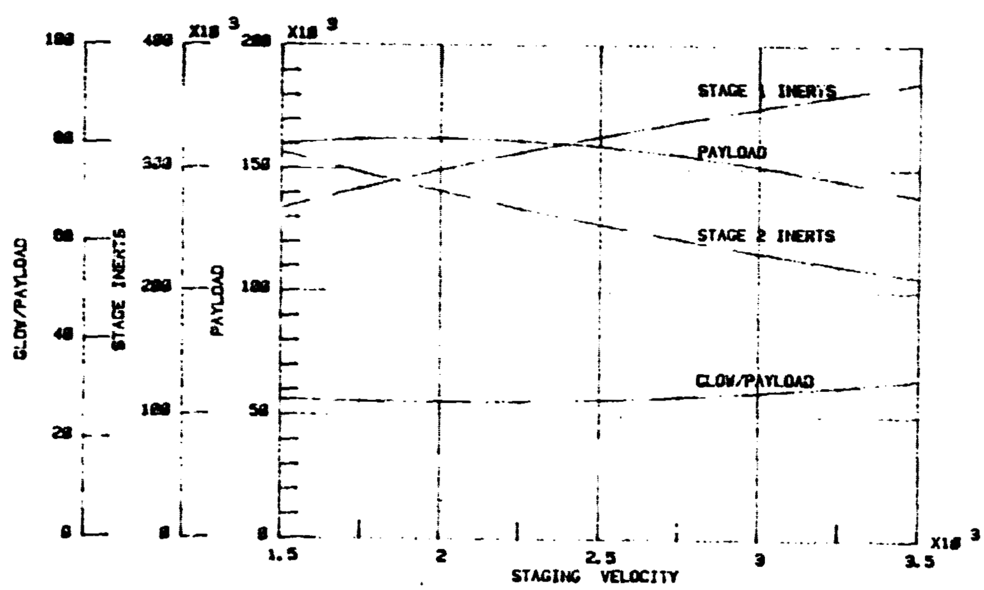

Figure 2.1-5. Series Burn–Glow = 4.5E6

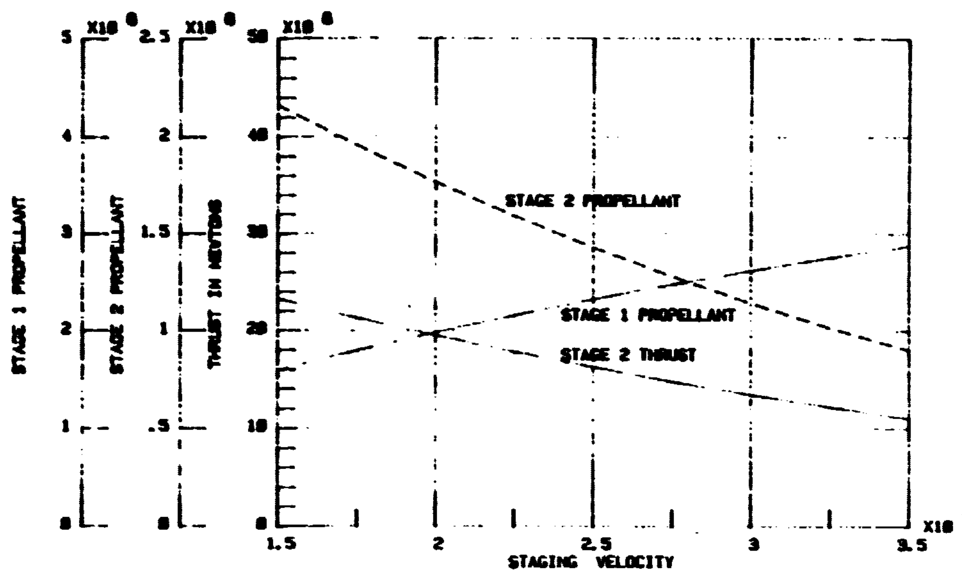

Figure 2.1-6. Series Burn—Glow = 4.5E6

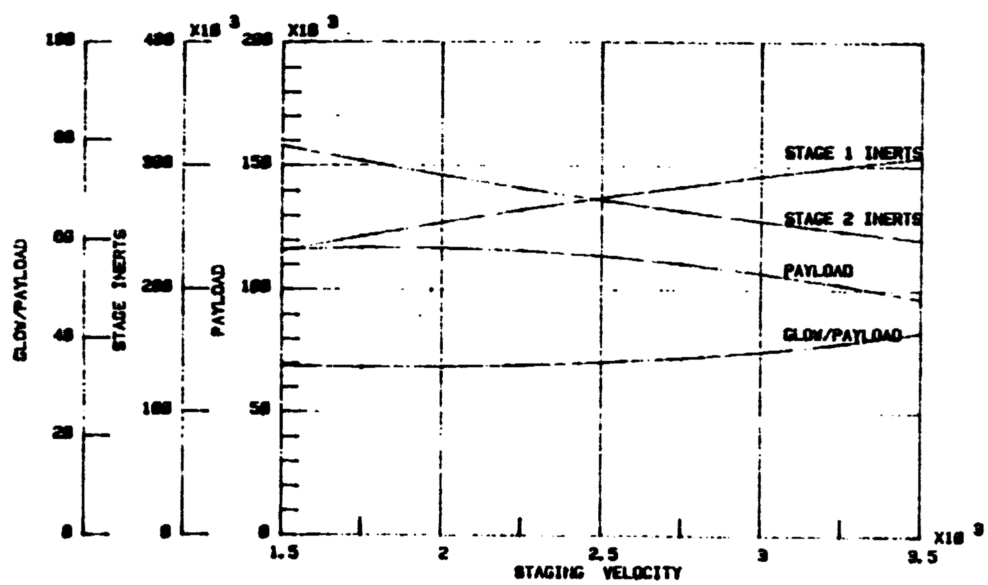

Figure 2.1-7. Parallel Burn—Glow = 4E6

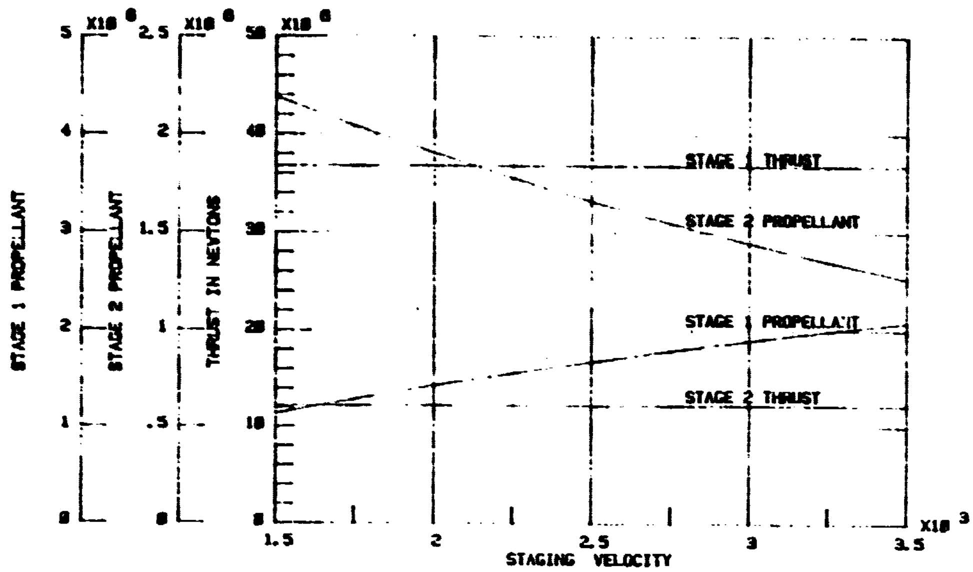

Figure 2.1-8. Parallel Burn—Glow - 4E6

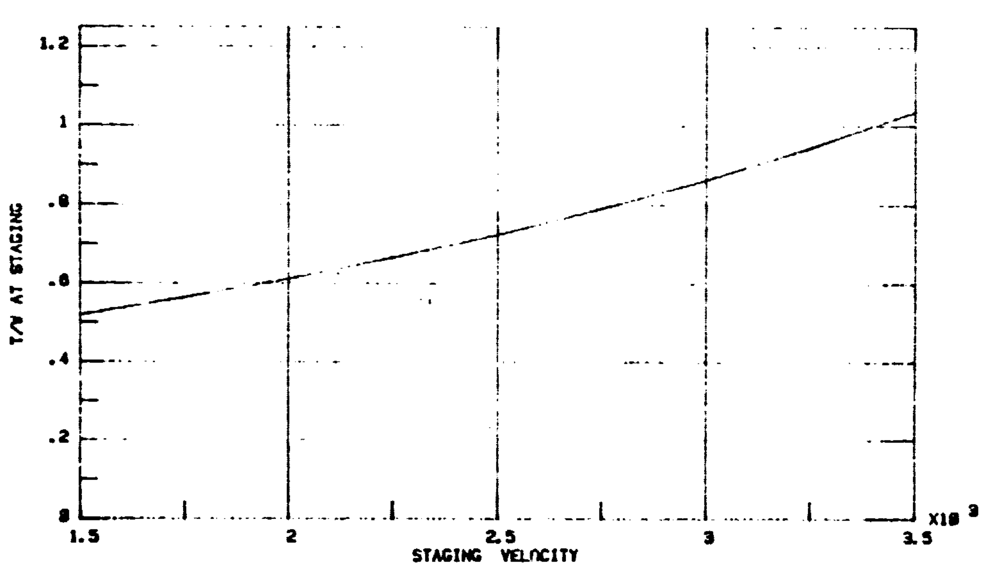

Figure 2.1-9. Parallel Burn—Glow - 4E6

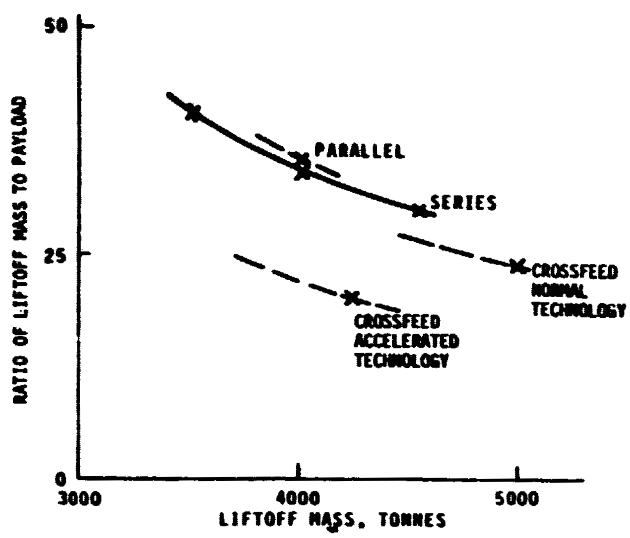

Figure 2.1-10. Mass Trending

Orbit eccentricity = e = $\left\{1 - \frac{r(2a-r)}{a^2(1+\tan^2\gamma)}\right\}^{1/2}$

where γ is inertial path angle. Our trajectory code gives only relative path angle, but both relative and inertial velocities. γ_I is less than γ_R by an amount

$$\Delta\gamma = \sin^{-1}\left\{\frac{V_0^2}{V_I^2} - \frac{(V_I^2 - V_R^2 - V_0^2)^2}{4V_I^2V_R^2}\right\}^{1/2}$$

where V_o is the velocity of Earth rotation, $\cong$ 407 m/s at KSC.

Flyback angle: $\alpha = 2(\pi - \Theta)$

where $\Theta = \cos^{-1}\left\{\frac{1}{e}\left[\frac{a(1-e^2)}{r} - 1\right]\right\}$

Flyback range = $r_\oplus \alpha$ where $r_\oplus$ is radius of Earth

This algorithm is plotted parametrically in Figure 2.1-11. The downrange distance of the staging point must be added to get total flyback range. Since range varies appreciably with path angle, trajectory depression to reduce flyback range may be an important consideration. This was to be investigated later by trajectory analyses.

2.1.3 Configuration Options and Selection

The configuration options examined included parallel and series burns vehicles. By prior agreement with JSC, the series burn vehicles did not consider crossfeed (supplying orbiter engines from booster tanks during mated flight). The advantage and disadvantages of crossfeed may be noted.

Advantage

o Orbiter propellant fraction is improved since the orbiter tanks need not accommodate orbiter engine propellants consumed during mated flight. The equivalent tankage inert mass is carried by the booster, where its effect on payload is 1/4 to 1/6 that of orbiter inert mass.

Disadvantages

(1) Propellant flow to orbiter engines must be "handed off" from the booster to the orbiter just prior to staging without interrupting orbiter engine operation;

(2) The booster must be configured to contain three propellants, i.e., O_2, CH_4 (or other hydrocarbon) and H_2.

(3) At staging, large-diameter propellant delivery lines between the booster and orbiter must be disconnected safely; if these lines penetrate a heat shield, protective doors must be closed. (This problem, of course, exists in separating the external tank from the space shuttle orbiter). If both stages are reusable, there is a problem of protruding lines, presumably from the booster. If the lines cannot be retracted (this would require large-diameter flex joints) it may be necessary to employ a jettisonable line section.

Three configurations were examined: a series-burn option, and two parallel burn options, belly-to-belly and back-to-back. These are shown in Figures 2.1-12 through 2.1-14. The series-burn design employs a "flower-petal" nose of six triangular struts that support the upper stage, each covered by a partial external fairing. After stage separation, the flower petal elements are retracted by actuators to form a smoothly-faired nose. With the petals open, flow paths exist to allow the second stage engine start sequence to be initiated prior to separation.

The belly-to-belly parallel burn configuration places the wings close together. This may reduce transonic drag, but structural connections penetrate the heat shields of both stages. The back-to-back option eliminates heat shield penetrations.

The series-burn option was selected for more detailed analysis. Rationale was as follows:

- o The series-burn vehicle has slightly better performance - 120 tonnes compared to 100 tonnes;
- o Stage separation is simpler; for parallel burn systems, the orbiter thrust after booster cutoff tends to push the stages together rather than push them apart;
- o Boost aerodynamics is simpler; the booster wing is in the orbiter wing wake rather than in an interfering location.
- o Ground handling is expected to be simpler.
- o The booster is more adaptable to use as a shuttle booster.
- o Mated vehicle propulsion tests are not needed to qualify the boost phase propulsion system.
- o Load paths and structural dynamics are simpler.

The principal disadvantage of series burn is the higher boost thrust required -about 1800K, per engine versus 1450K.

The series-burn stack height is commensurate with that of Saturn V, indicating that present facilities can be used in the developmental phase. The operational, high-launch-rate, ground handing system will probably move the empty vehicles on their own landing gear, mate in the horizontal position at the launch pad, and use a strong-back tilt-up launcher.

2.2 VEHICLE ANALYSIS

The following discussion presents results of analyses of the series-burn vehicle.

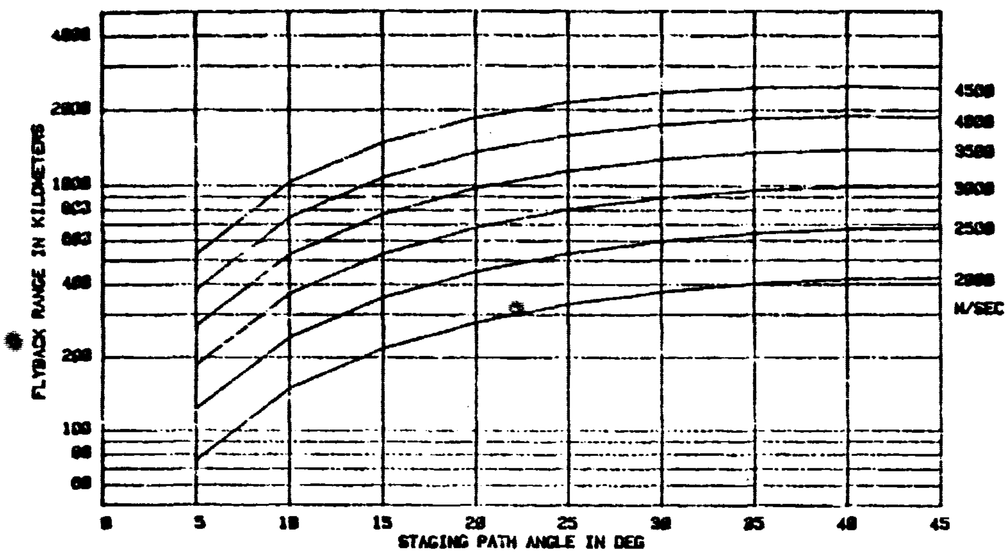

Figure 2.1-11. Flyback Range Parametrics

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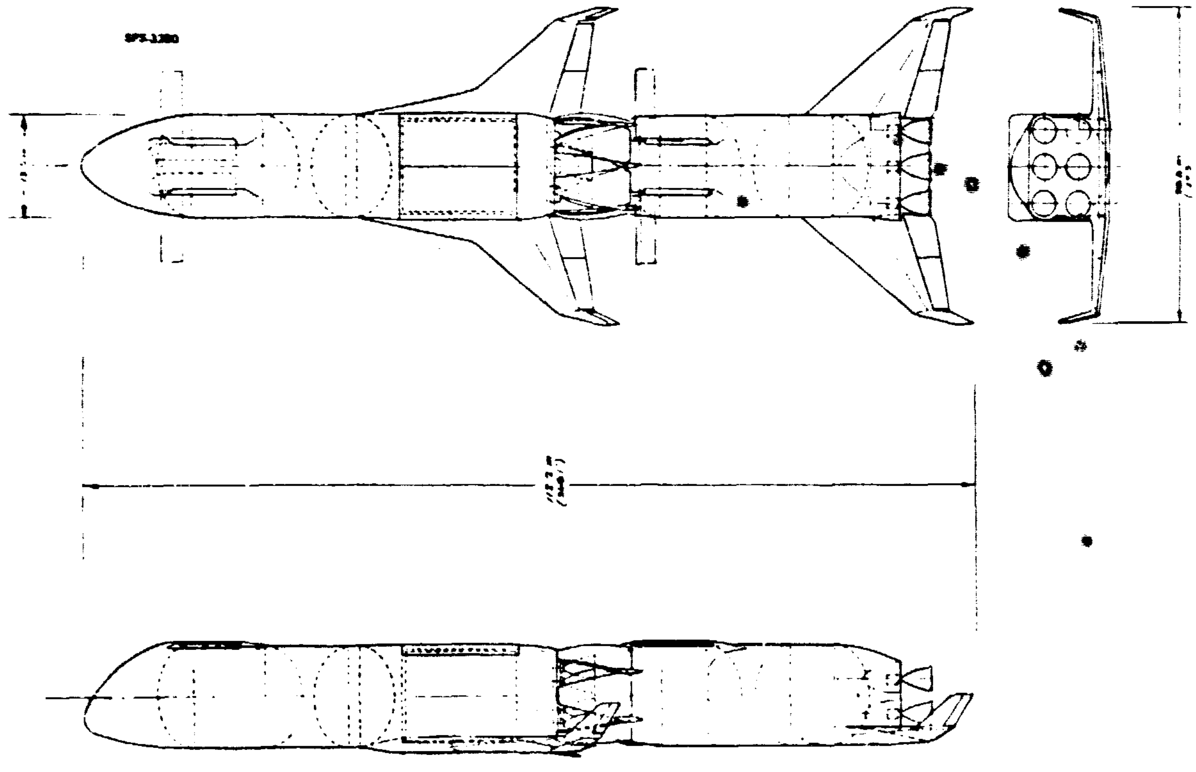

Figure 2.1-12. Series Burn HLLV

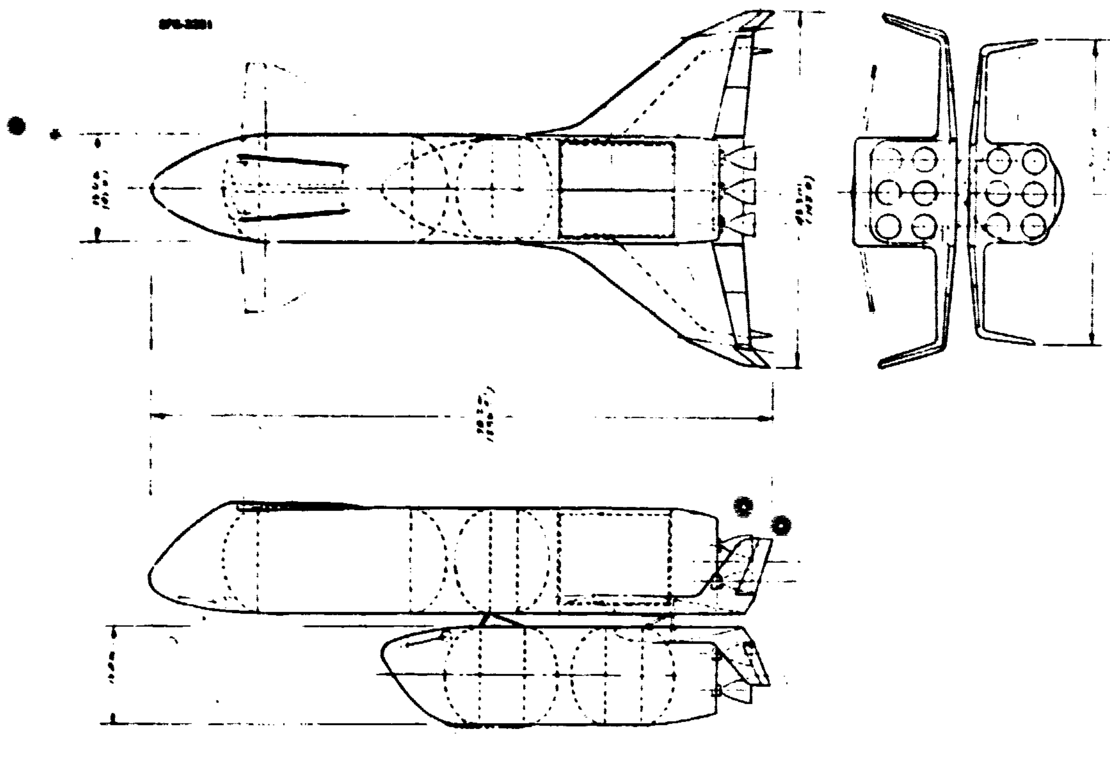

Figure 2.1-13. Parallel Burn HLLV

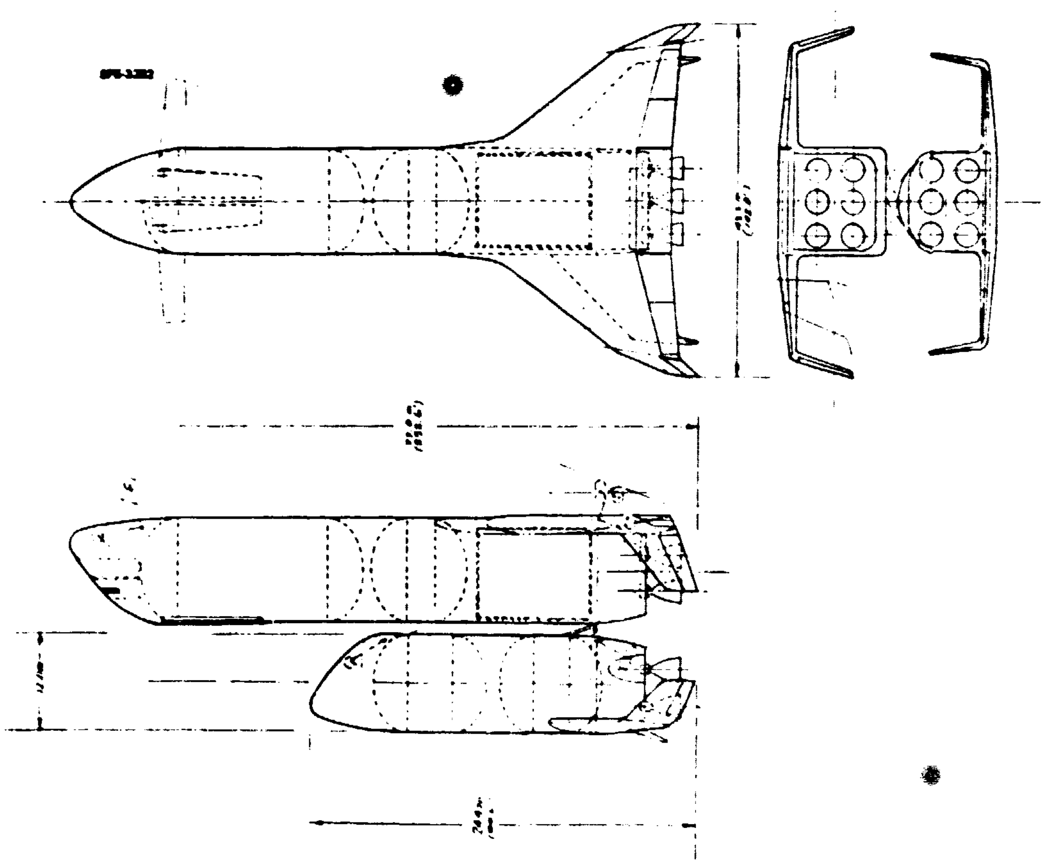

Figure 2.1-14. Parallel Burn HLLV

2.2.1 Trajectory Analyses and Vehicle Optimization

The vehicle launch trajectory employs zero-lift "gravity-turn' boost trajectory followed by a roughly optimized second stage trajectory. Injection conditions are 90km altitude, due east, with injection velocity appropriate to coast to 477km altitude.

Shortly after liftoff, the mated vehicle (under booster thrust) executes a slight "tilt" away from vertical flight, in the downrange direction. This initiates the "gravity turn." The amount of tilt sets the staging conditions. With a fixed amount of boost propellant, more tilt (a) reduces staging altitude; (b) reduces staging path angle; (c) increases relative velocity at staging. It is intuitively logical that there should be an optimal tilt; this is indeed true. The objective is to maximize injected mass (the sum of second stage inert mass and payload). Figure 2.2-1 shows variation in staging parameters and in injected mass as a function of tilt angle. Figures 2.2-2 and 2.2-3 show the characteristics of a preliminary reference trajectory with near-optimal characteristics.

Final selection of a reference trajectory requires evaluation of flyback range effects. For any flyback range, there will be an optimal booster wing area. Increasing wing area increases the flyback cruise L/D, decreasing both installed thrust and flyback fuel. Since increasing wing area reaches a point of diminishing returns, i.e., further increases in area add little to L/D, whereas wing mass increases nearly linearly with area, it is apparent that an optimal area must exist (for any given flyback range). Since booster inerts affect payload (1 kg of booster inerts is worth roughly 1/6 kg payload) there is a joint optimum among staging conditions and booster wing area. These optimizations are nearly decoupled, however, because of the sharpness of the optimum of tilt (= staging conditions). The flyback range at optimal staging conditions will be between 250 and 300 km. Over this range the optimal wing area will change little. Consequently, our analysis assumed these optima to be entirely decoupled.

2.2.2 Aerodynamics

A further parametric study was conducted to select the reference wing area. Wing area was dictated by landing speed with a desire to maintain landing speed at no more than 165 knots. The result was a selection of a reference wing area of 8200 ft^2 with a canard for subsonic trim, as shown in Figure 2.2-4.

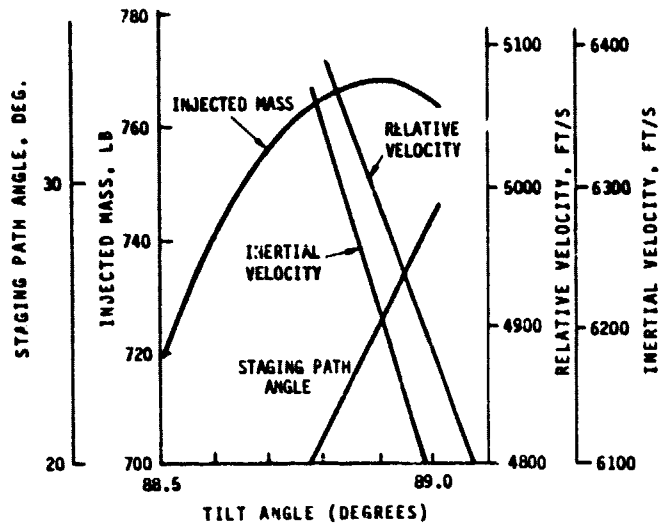

Figure 2.2-1. Staging Point Variation and Injected Mass

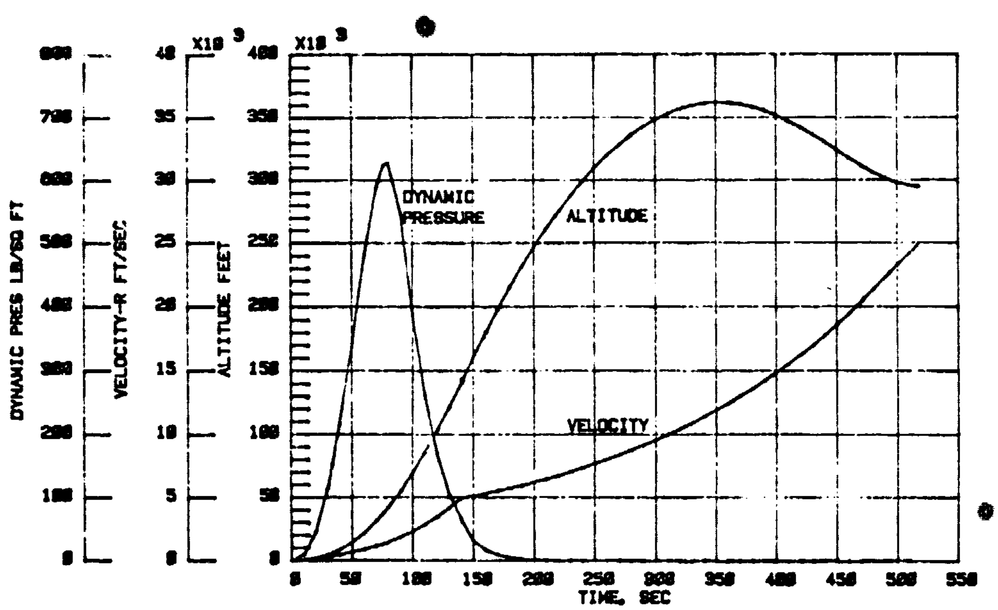

Figure 2.2-2. Small HLLV Reference Trajectory

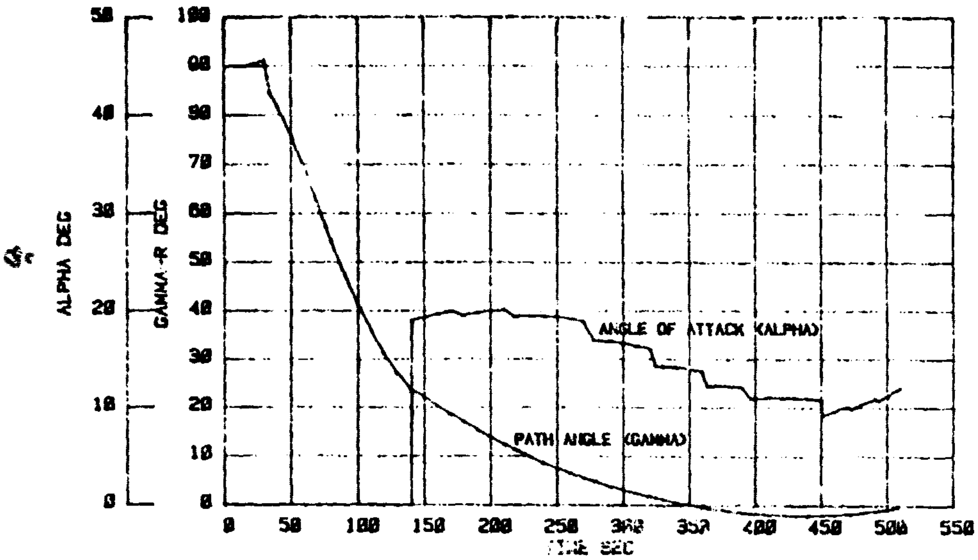

Figure 2.2-3. Small HLLV Reference Trajectory

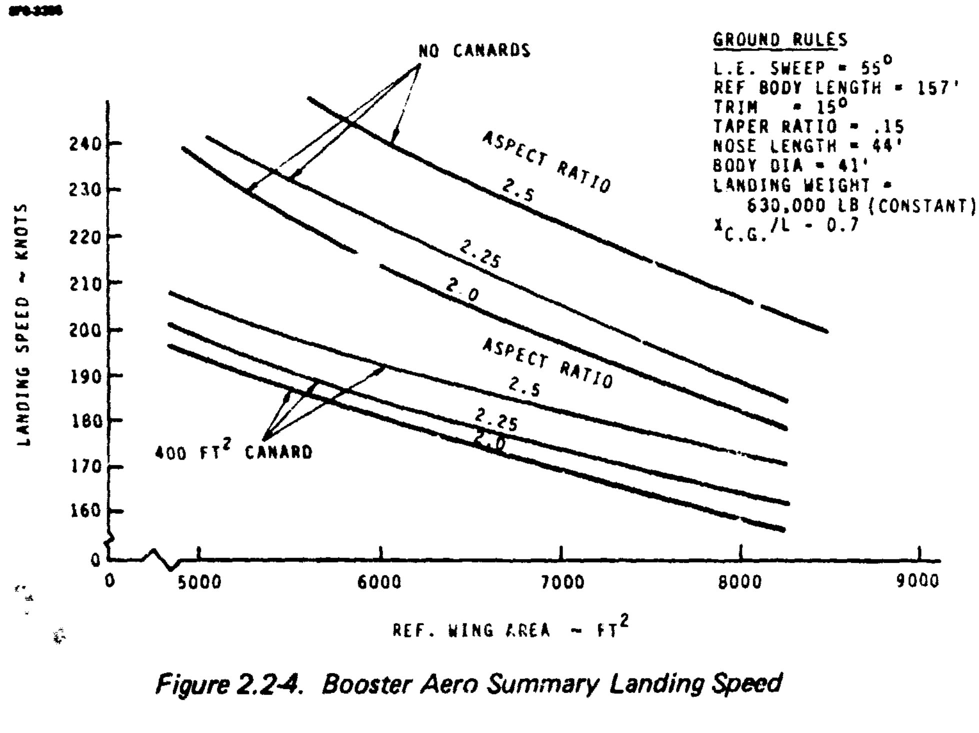

Figure 2.2-4. Booster Aero Summary Landing Speed

A hypersonic trim investigation, summarized in Figure 2.2-5, showed that the vehicle could be trimmed between 30 and 40 degrees angle of attack with reasonable aileron deflections.

The orbiter wing area was also selected for landing speed of 165 knots. Again, a canard was used for subsonic trim to avoid large wing areas. The landing speed parametrics are shown in Figure 2.2-6.

Table 2.2-1 summarize the results of the aerodynamics investigations.

As a result of the aerodynamics investigation, the vehicle wings were resized.

Illustrated in Figure 2.2-7 are the revised wing area as compared to the original wing areas, shown on the original configuration. Revised wing areas are shown as dotted lines.

2.2.3 Selected Configuration

The small HLLV final configuration is shown in Figure 2.2-8. The orbiter includes a swept-back delta wing with a small subsonic foldout canard. The payload bay is aft of the propellant tanks and is 11 metres square by 14 metres long. The orbiter uses six space shuttle main engines with extended exit bells. Four of the six engines are gimbaled; the center two are fixed. The upper stage also uses a small yaw ventral for head-end steering to improve controllability in yaw.

The vehicles are control configured in yaw, thus eliminating the large vertical tail. Elimination of the vertical tail assists in balancing the vehicle and makes practical an aft payload bay on the orbiter. The booster employs a "flower-petal" opening nose with a truss structure as an interstage structure. This approach avoids expendible interstage hardware and allows the second stage engine start sequence to be initiated during the first stage tail-off as the open nose allows room for gas venting during the start sequence. After stage separation, a simple hinged actuator mechanism closes the nose to a streamlined, aerodynamic configuration.

The booster employs six oxygen-methane engines of approximately 1835 K/lb thrusts. Four high thrust air-breather engines are mounted on top of the wings for fly-back. The air-breather engine inlets are closed by a blow-off cover until subsonic transition at which time the engines undergo start sequence. Engine location was selected to avoid flow attachment to either the wing or the body as a flow attachment will result in higher drag during the fly-back.

2.2.4 Mass Properties

Table 2.2-2 presents the mass statement for the small HLLV, based on the final configuration. The estimated payload based on the detailed mass statement is 126 metric tons as compared to a parametric figure of 120 metric tons.

2.2.5 HLLV Fleet Size Scenario

The SPS transportation and construction system interrelated transportation operations scenario material presented in the reference system description report from Phase II has been incorporated into software so that trade studies can be run. Shown in Table 2.2-3 is the HLLV fleet scenario for the small HLLV. Note the increased numbers of flights and the increased production rate. These scenario results provided the basics for cost analyses.

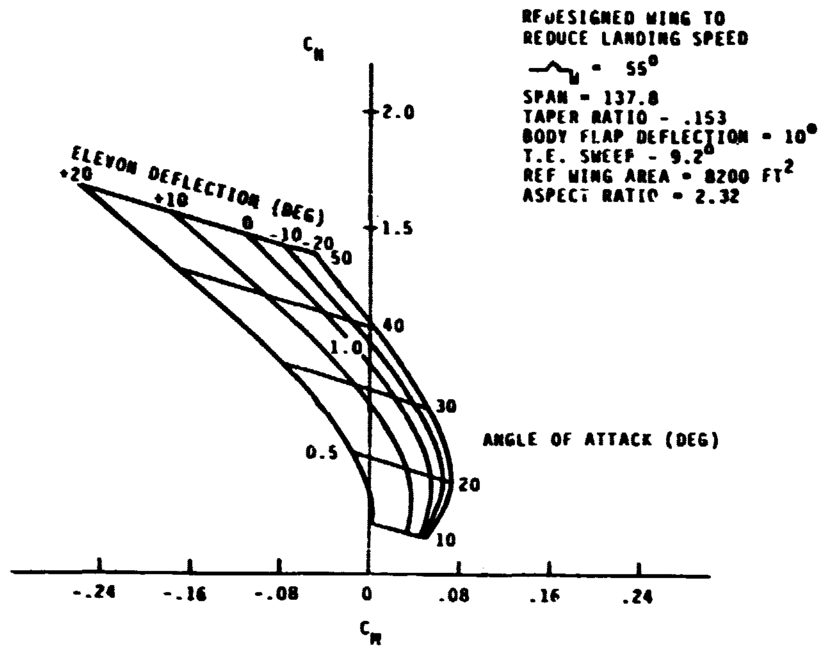

Figure 2.2-5. Booster Aero Summary Hypersonic Trim

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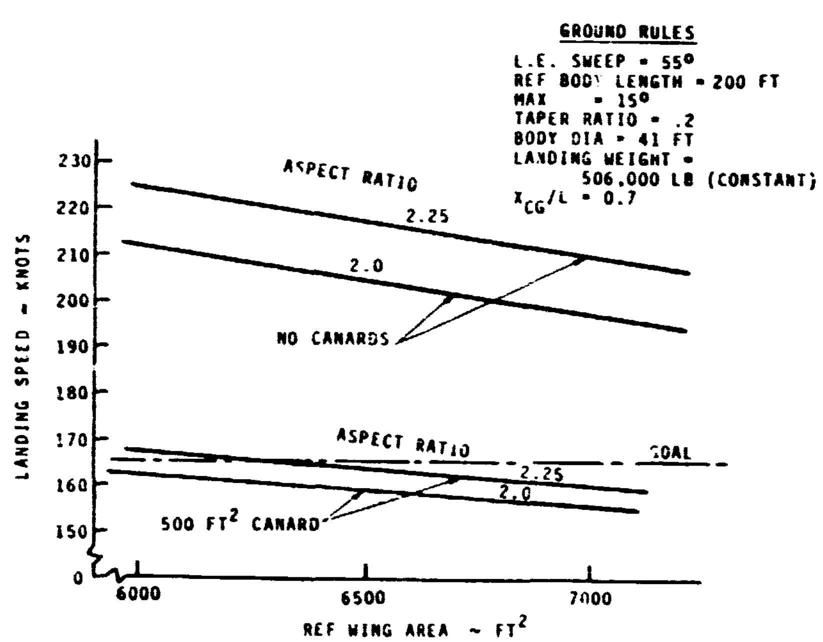

Figure 2.2-6. Effect of REF Wing Area and Aspect Ratio on SPS Orbiter Landing Speed

TABLE 2.2-1

SUMMARY OF RESULTS OF INITIAL ITERATION ON SPS BOOSTER/ORBITER AERODYNAMICS

- **BOOSTER**
  - INITIALLY DEFINED CONDITIONS
    - WEIGHT AT START OF FLYBACK

      320 TONNES = 704,000 LBS
    - FLYBACK RANGE

      250 KM + 20 MINUTES RESERVE
    - C.G.

      $X_{C.G.}$/BODY LENGTH = 0.7
    - DRAWING OF CONFIGURATION
  - ADDITIONAL CONDITIONS DEFINED
    - LANDING
      - ANGLE OF ATTACK = 15^{o} MAX
      - SPEED = 165 KTS MAX.
    - HYPERSONIC TRIM
      - TRIM BETWEEN 30^{o} & 50^{o} ANGLE OF ATTACK
      - TRIM WITHOUT POSITIVE ELEVON DEFLECTION
  - RESULTS
    - LANDING SPECIFICATIONS CONTROL WING AREA
      - ORIGINAL WING REF AREA FROM DWG = 6000 FT^2
      - REQUIRED WING REF AREA = 8000 FT^2

TABLE 2.2-1 (Continued)

- FLYBACK
  - o C_{D_O} .032 (BASED ON WING REF AREA)
  - o ASSUME FLYBACK OCCURS AT $L/D)_{MAX}$ AND 10,000 FT ALTITUDE
  - o ASSUME TSFC = 0.8 FOR FLYBACK ENGINES
  - o CONCLUSIONS
    - o $(L/D)_{MAX}$ = 6.73
    - o 67,000 LB FUEL REQD. (INCLUDING 20 MINUTES RESERVES)
    - o VELOCITY = 500 KM/HR
    - o WING LOADING AT START OF FLYBACK 86 LB/FT^2
    - o 105,000 LB THRUST REQD. AT START OF FLYBACK
- HYPERSONIC TRIM

  BOOSTER WILL TRIM AT 35° ANGLE OF ATTACK
  WITH 0° ELEVON DEFLECTION
- RECOMMENDED WING/CANARD DESIGN

  REF AREA = 8200 FT^2
  ASPECT RATIO = 2.32
  L.E. SWEEP = 55°
  TAPER RATIO = .15
  T.E. SWEEP = 9.2°
  CANARD AREA = 400 FT^2
  LANDING TRIM CL = .83
  ELEVON/WING AREA = .12
  ELEVON DEFLECTION = 7.6°

TABLE 2.2-1 (Continued)

- **ORBITER**
  - INITIALLY DEFINED CONDITIONS
    - LANDING WEIGHT = 230 TONNES 506,000 LB
    - X_{CG}/BODY LENGTH = 0.7
    - DRAWING OF CONFIGURATION
  - ADDITIONAL CONDITIONS DEFINED
    - LANDING ANGLE OF ATTACK = 15^{o} (MAX)
    - LANDING SPEED = 165 KTS (MAX)
  - RESULTS
    - ORIGINAL WING REF AREA OF 5600 FT^2 WAS A LITTLE LOW FOR LANDING
    - RECOMMENDED WING/CANARD CONFIGURATION
      - REF WING AREA = 6180 FT^2
      - REF WING ASPECT RATIO = 2.25
      - REF WING TAPER RATIO = .186
      - WING L.E. SWEEP - 55^{o}
      - WING T.E. SWEEP = 12^{o}
      - CANARD AREA = 500 FT^2
      - LANDING TRIM CL = 0.88
      - ELEVON/WING RATIO = .12
      - ELEVON DEFLECTION = 11^{o}

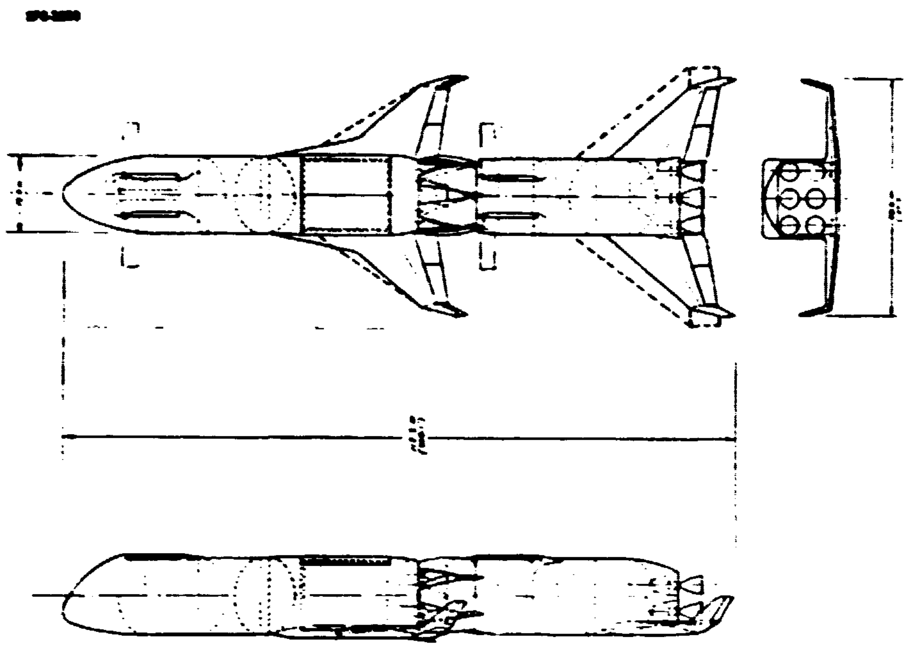

Figure 2.2-7. Small HLLV–Wing Resize

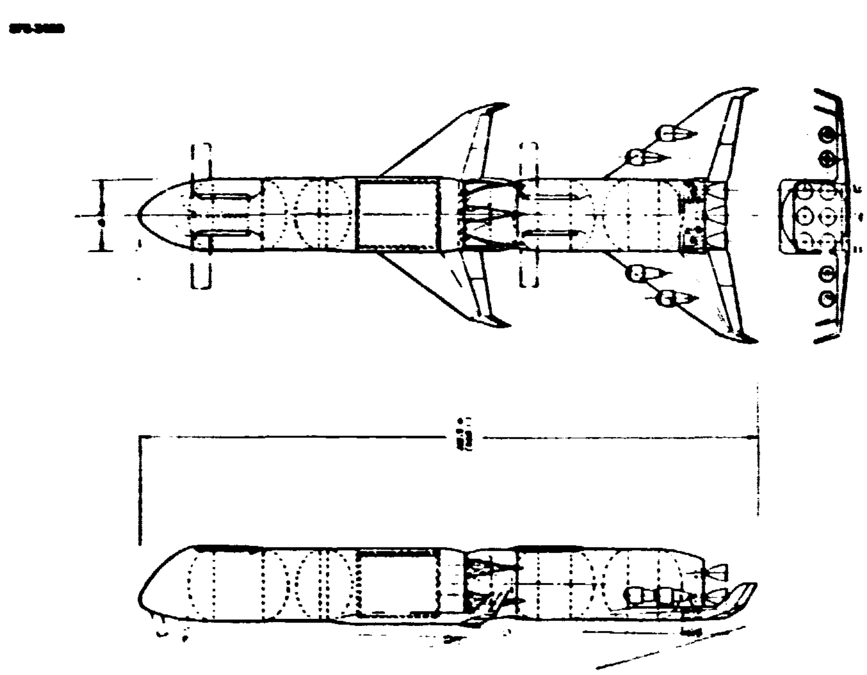

Figure 2.2-8. Small HLLV Updated Configuration

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TABLE 2.2-2

SMALL HLLV MASS PROPERTIES

BOOSTER	KG	LBM
STRUCTURE-AEROSURFACES	28,235	62,245
WING	25,509	56,236
CANARD	1,452	3,200
TIPLETS	1,020	2,249
YAW VENTRAL	254	560
STRUCTURE - BODY & TANKS	69,107	152,357
NOSE	9,761	21,519
NOSE GEAR SUPPORT	693	1,528
METHANE TANK	9,684	21,349
OXYGEN TANK	13,610	30,006
INTERTANK	10,592	23,353
AFT BAY & FAIRINGS	10,513	23,178
THRUST STRUCTURE	8,130	17,924
BODY FLAP	1,860	4,100
FAIRINGS	4,264	9,400
TPS	0	0
MECHANISMS	9,043	19,936
LANDING GEAR	8,090	17,836
DRAG DEVICE	953	2,100
MAIN PROPULSION	68,750	151,596
ROCKET ENGINES	50,000	110,229
ENGINE ACCESSORIES	6,250	13,779
PROPELLANT SYSTEMS	12,500	27,588
AUXILIARY PROPULSION	30,615	67,495
FLYBACK ENGINES	25,000	55,115
FUEL SYSTEM	3,039	6,700

TABLE 2.2-2 (Continued)

SMALL HLLV MASS PROPERTIES

BOOSTER-(CON'T)	KG	LBM
RCS	2,576	5,680
SUBSYSTEMS	7,804	17,205
AUXILIARY POWER	703	1,550
ELEC. CONV & DISTR.	2,667	5,880
FLT CONTROL ACTUATION	2,073	4,570
FLIGHT CONTROL SYSTEM	1,111	2,450
AVIONICS	1,000	2,205
EC/LSS	250	550
GROWTH	21,355	47,083
TOTAL DRY	234,909	517,917
FLUIDS	61,466	135,506
BIAS PROPELLANT	11,300	24,911
PRESSURANT	11,300	24,911
RESIDUALS & TRAPPED	8,475	18,684
FLYBACK FUEL	30,391	67,000
NET INERTS	296,375	653,423
IMPULSE PROPELLANT	2,260,000	4,982,396
BOOSTER LIFTOFF MASS	2,556,375	5,635,819

ORBITER	KG	LBM
STRUCTURE-AEROSURFACES	22,552	49,720
WING	20,135	44,390
CANARD	1,560	3,440
TIPLETS	635	1,400
YAW VENTRAL	222	490

TABLE 2.2-2 (Continued)

SMALL HLLV MASS PROPERTIES

ORBITER (CON'T)	KG	LBM
STRUCTURE-BODY & TANKS	66.328	146,211
NOSE	2,440	5,380
NOSE GEAR SUPPORT	529	1,166
LH_2 TANK	10,928	24,093
LO_2 TANK	11,719	25,835
INTERTANK	6,231	13,737
PAYLOAD BAY BODY SECTION	10,282	22,668
PAYLOAD BAY DOORS	2,255	4,971
AFT BODY	10,979	24,204
THRUST STRUCTURE	3,390	7,473
BODY FLAP	2,270	5,000
FAIRINGS	2,137	4,700
CREW CAB STRUCTURE	3,168	6,984
INDUCED THERMAL PROTECTION	19,923	43,922
WING RSI	4,799	10,580
BODY RSI	10,136	22,345
TANK SIDEWALL PANELS	1,571	3,465
WING TIPLETS RSI	386	850
LH_2 INTERNAL INSULATION	2,169	4,782
PROPELLANT PURGE, VENT, & DRAIN	862	1,900
MECHANISMS	7,198	15,869
LANDING GEAR	6,439	14,196
DRAG DEVICE	759	1,673
MAIN PROPULSION	31,694	69,873
SSME's	19,336	42,630

TABLE 2.2-2 (Continued)

SMALL HLLV MASS PROPERTIES

ORBITER (CON'T)	KG	LBM
ACCESSORIES	2,077	4,580
AFT BODY PROPELLANT SYSTEM	7,008	15,450
DELIVERY LINES & PROP. MGT	3,273	7,213
AUXILIARY PROPULSION	4,090	9,018
OMS PROPULSION SYS (DRY)	2,548	5,618
RCS PROPULSION SYS (DRY)	1,542	3,400
SUBSYSTEMS	9,960	21,958
FLIGHT CONTROL	1,270	2,800
AVIONICS	1,978	4,360
EC/LSS	1,339	2,952
ELECTRIC POWER	5,373	11,846
CREW & PAYLOAD ACCOMMODATIONS	3,652	8,053
PERSONNEL PROVISIONS	305	674
FURNISHINGS	411	907
PAYLOAD PROVISIONS	1,380	3,042
CREW & ACCESSORIES	1,556	3,430
GROWTH	14,519	32,009
TOTAL DRY WITH CREW	179,916	396,633
FLUIDS & GASES	41,734	92,008
OMS PROPELLANT	28,263	62,309
OMS RESERVES & RESIDUALS	2,826	6,231
FUEL CELL REACTANT	254	560
TRAPPED MAIN PROPELLANT & PRESSURANT	10,391	22,908

TABLE 2.2-2 (Continued)

SMALL HLLV MASS PROPERTIES

ORBITER (CON'T)	KG	LBM
TOTAL INERTS	221,650	488,641
ASCENT PAYLOAD	126,260	278,359
TOTAL ORBITER INJECTED	347,910	767,000
INTEGRATED VEHICLE		
IMPULSE PROPELLANT	1,130,000	2,491,198
ORBITER AT LIFTOFF	1,477,910	3,258,198
BOOSTER AT LIFTOFF	2,556,375	5,635,819
VEHICLE AT LIFTOFF	4,034,285	8,894,017

Vehicle quantities were derived from the scenario data in Table 2.2-3. The scenario analysis establishes the number of vehicles required for the initial fleet. Spares were added to this. Engines and auxiliary propulsion were independently estimated. Since the engines follow a different learning curve than the airframes, it is necessary to discretely estimate engine costs. The scenario results also determine the number of new vehicles required for life cycle operations. An additional set of equivalent vehicles is required to maintain spares and maintenance. Table 2.2-4 summarizes the results of this analysis. The figures used were based on the same assumptions as used to cost the reference HLLV.

Table 2.2-3. Small HLLV Transportation Scenario

HLLV FLEET

PROGRAM YEAR	NO. OF SPS'S	TOTAL PAYLOAD CONSTR	TOTAL PAYLOAD MAINT	HLLV FLIGHTS CONSTR	HLLV FLIGHTS MAINT	TOTAL FLIGHTS	BOOSTERS REQD	ORBITERS REQD	BOOSTER BUYS	ORBITER BUYS	CREW CAP. REQD
1	0	27364	0	69.7	2.0	71.7	1	2	2.2	3.2	2.8
2	0	6149	154	63.0	0.0	63.0	1	1	0.2	0.2	3.2
3	1	35027	2877	330.5	4.4	334.9	4	5	4.1	4.1	1.6
4	2	102624	6724	982.9	6.7	989.5	12	15	11.3	13.3	1.6
5	4	135515	8300	1308.8	19.3	1328.1	16	20	8.4	9.4	1.7
6	6	134723	7768	1308.6	34.3	1342.9	16	20	4.5	4.5	1.8
7	8	134741	9387	1308.7	49.4	1358.1	16	20	4.5	4.5	1.9
8	10	134757	10684	1308.8	61.0	1369.9	16	20	4.6	4.6	2.0
9	12	134774	12196	1309.0	77.0	1385.9	16	21	4.6	5.6	2.1
10	14	134792	13923	1309.1	91.2	1400.2	17	21	5.7	4.7	2.2
11	16	134917	15712	1309.3	103.9	1413.2	17	21	4.7	4.7	2.3
12	18	134825	16732	1309.3	118.7	1428.1	17	21	4.8	4.8	2.4
13	20	134843	18265	1309.5	130.8	1440.3	17	21	4.8	4.8	2.4
14	22	134862	19991	1309.6	149.0	1458.6	17	22	4.9	5.9	2.5
15	24	134877	21289	1309.7	160.7	1470.4	17	22	4.9	4.9	2.6
16	26	134895	22908	1309.8	173.7	1483.6	18	22	5.9	4.9	2.7
17	28	135021	24912	1310.0	190.8	1500.8	18	22	5.0	5.0	2.8
18	30	134928	25825	1310.1	202.4	1512.5	18	22	5.0	5.0	2.9
19	32	134946	27337	1310.2	214.3	1524.6	18	23	5.1	6.1	2.9
20	34	134963	28956	1310.3	231.4	1541.7	18	23	5.1	5.1	3.0
21	36	134982	30596	1310.5	246.6	1557.1	18	23	5.2	5.2	3.1
22	38	134998	31894	1310.6	258.3	1568.9	19	23	[illegible].2	5.2	3.2
23	40	135015	33406	1310.7	272.2	1582.9	19	23	[illegible]3	5.3	3.2
24	42	135142	35517	1310.9	288.4	1599.3	19	24	[illegible].3	6.3	3.3
25	44	135050	36537	1311.0	301.2	1612.1	19	24	5.4	5.4	3.4
26	46	135066	37942	1311.1	315.9	1627.1	19	24	5.4	5.4	3.4
27	48	135083	39453	1311.2	327.9	1639.1	19	24	5.5	5.5	3.5
28	50	135102	41180	1311.4	344.1	1655.4	20	25	6.5	6.5	3.6
29	52	135118	42498	1311.5	355.9	1667.4	20	25	5.6	5.6	3.7
30	54	135244	44503	1311.7	372.9	1684.6	20	25	5.6	5.6	3.7
31	56	135154	45737	1311.7	386.0	1697.7	20	25	5.7	5.7	3.8
32	58	135170	47034	1311.9	399.6	1711.5	20	25	5.7	5.7	3.8
33	60	135187	48546	1312.0	411.6	1723.5	20	26	5.7	6.7	3.9

TOTAL FLIGHTS: 45744.7
TOTAL BOOSTERS BOUGHT: 173.482   TOTAL ORBITERS BOUGHT: 179.482

Table 2.2-4. Vehicle Quantities

SPS-3417

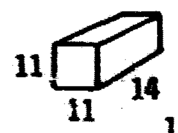
INITIAL FLEET & SPARES	BOOSTER	ORBITER
AIRFRAME	17	22
MAIN ENGINE	102	133
AUX. PROPULSION	70	22
LIFE CYCLE		
NEW VEHICLES		
AIRFRAME	173	179
MAIN ENGINE	1041	1077
AUX. PROPULSION	694	179
SPARES & MAINTENANCE		
AIRFRAME	174	174
MAIN ENGINE	2744	2744
AUX. PROPULSION	101	174

2.3 THE EFFECTS OF A SMALL HLLV ON PAYLOAD PACKAGING, SPS CONFIGURATION, GROUND AND SPACE FACILITIES, AND OPERATIONS

2.3.1 Small HLLV Packaging Parameters

The nominal small HLLV payload parameters that were given are as follows:

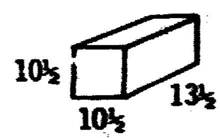
Cargo Bay Envelope



Payload Mass 120mT

Following the guidelines established in previous packaging analyses (Reference: Section 5 in Reference 9), we have discounted these parameters to allow for packaging and pallets. The working parameters become the following:

Max. envelope of components



Max. payload mass (without packaging) 108mT

Table 2.3-1 lists the total payload that needs to be delivered to LEO for each year of the SPS commercial program. This total payload includes components, spare parts, crew supplies and propellants used at both LEO and GEO. This table also lists the corresponding number of mass-limited launches required per year and per day to deliver this payload.

2.3.2 Effects on SPS Program Elements

The constraints identified in the previous section were used to define the effects on the various SPS program elements. Table 2.3-2 lists the program elements directly or indirectly effected by having a smaller HLLV. (The reader should refer to Reference 7 as this table is examined.) Elements not identified in this table are not affected.

The interactions of these effects are more clearly shown in Figure 2.3-1. It is seen that there are eight primary effects. It should be evident from this map that if any of the 8 primary effects can be alleviated, the secondary effects linked to them can also be eliminated. The possibilities for alleviating the primary effects are discussed in Table 2.3-3.

As a part of this analysis, the personnel OTV was reconfigured to fit the shorter payload bay. The revised OTV concept is shown in Figures 2.3-2, 2.3-3 and 2.3-4.

2.3.2.1 Supporting Analyses

There were three supporting analyses that were conducted to derive some of the data shown in the preceding tables. These were a cargo packaging analysis, a GEO Base effects analysis, and alternative launch and recovery site concepts analysis.

2.3.2.1.1 Cargo Packaging Analysis

The primary objective of the cargo packaging analysis was to determine the configurations of the primary payloads for the small HLLV.

The cargo packaging data developed in Phase II of this study were used as the reference (see Table 5-1 in Section 5.0—Cargo Packaging in Reference 9). These data were

TABLE 2.3-1
THEORETICAL QUANTITY OF MASS-LIMITED LAUNCHES

SPS PROGRAM YR	TOTAL PAYLOAD (MT) [1]	THEORETICAL TOTAL NO. OF LAUNCHES (MASS-LIMITED) [2]	NO. OF LAUNCHES [3] PER DAY	NO. OF LAUNCHES [3] PER WEEK
1	15059	140	.38	2.66
2	17048	158	.43	3.01
3	47095	437	1.20	8.4
4	107633	997	2.73	19.11
5	138549	1283	3.52	24.64
6	137065	1270	3.48	24.36
7	138990	1287	3.55	24.85
8	140104	1297	3.55	24.85
9	141661	1312	3.59	25.13
10	155249	1438	3.94	27.58
11	156457	1449	3.97	27.79
12	158804	1471	4.03	28.21
13	148352	1374	3.76	26.32
23	162564	1506	4.12	28.84
33	179013	1658	4.54	31.78

[1] Reference: D180-25461-2, Table 1.3-16 (p. 216)

[2] Based on 108 MT net payload per launch (120 MT payload capability discounted 10% to allow for packaging)

[3] Based on 7 day per week launch schedule

TABLE 2.3-2

EFFECTS OF THE SMALLER HLLV ON THE SPS PROGRAM

WBS	ITEM	DESCRIPTION OF EFFECT	MASS MT	COST [1] [2] $M
	Solar Power Satellite			
1.1.1.1.1	Primary Structure	o Redesign Type A Beam (the 12.7m beam) o Need battens every 7.5m	+196.7	+10.82 P
1.1.1.1.2	Catenary System	o Redesign catenary system to be compatible with the 7.5m wide solar array blankets	+11.264	+.536 P
1.1.1.3	Solar Blankets	o Redesign blankets to be 7.5m wide	0	0
		o Redesign cell string parallel and series interconnect scheme to alleviate need to interconnect 2 adjacent blankets		
1.1.1.3.1	Solar Cell Panels	o Revise panel size to be compatible with 7.5m blanket width	0	0
1.1.1.3.2	Interbay Jumpers	o Will have at least one more interbay jumpers per 15m and their associated hardware	+4.46	+.223 P
1.1.1.4.2	Acquisition Busses	o Revise acquisition bus configuration to accommodate 7.5 blankets	+19.8	+.73 P
1.1.4	Attitude Control and Stationkeeping	o The ion propulsion panel will have to be fabricated from 4 pieces instead of 2 pieces.	0	0
1.1.6.3	Power Distribution	o The electrical rotary joint assembly will have to be assembled from at least 4 large sub-assemblies instead of being delivered in one piece.	+.931	+4.0 P

[1] Cost Category Code: P = Production; I = Investment; Ø = Operations

[2] Cost of transporting additional mass has not been included.

TABLE 2.3-2 (Con't)

EFFECTS OF THE SMALLER HLLV ON THE SPS PROGRAM

WBS	ITEM	DESCRIPTION OF EFFECT	MASS MT	COST [1] [2] $M
	Geo Base			
1.2.1.1.2	Construction Equipment	o 12.7m Beam Machine will have to be revised for a redesigned Type A Beam (see WBS 1.1.1.1.1)	0	0
		o 30m Cherrypicker—Add 4 more of these to accommodate requirement to install twice as many solar array blankets per bay, and 2 more to assemble modular slip ring assembly.	25	+185.6 I
1.2.1.1.3	Cargo Handling and System	To accommodate smaller and more numerous cargo pallets:		
		o Cargo Tug Docking Ports—Add 2 docking systems	1	+2.6 I
		o Cargo Pallet Handling Jig—Add 2 units	1	+1.8 I
		o Transporters—Add 80 units (smaller size) in lieu of 20 large units	20	+74.2 I
1.2.1.1.4	Subassembly Factories	o Add a Electrical Rotary Joint Subassembly area and equipment (refer to WBS 1.1.6.3)	7.5	+48.8 I
		o Revice layout of thruster subassembly area	0	0
		o Beam and fitting subass'y revised to new Type A beam	0	0
1.2.1.1.5	Test/Checkout Facilities	o Add electrical rotary joint test facility, support equip., etc.	7.5	+48.8 I

[1] Cost Category Code: P = Production; I = Investment; Ø = Operations

[2] Cost of transporting additional mass has not been included.

TABLE 2.3-2 (Con't)

EFFECTS OF THE SMALLER HLLV ON THE SPS PROGRAM

WBS	ITEM		DESCRIPTION OF EFFECT	MASS MT	COST [1] [2] $M
1.2.1.1.6	SPS Maintenance Support Facilities	o	The KTM Refurbishment Facility will have to be redesigned to fit into the smaller crew modules (see WBS 1.2.1.2.2)	168 to 504	+1429 to 4288 I
		o	Need small crew habitats (see WBS 1.2.1.2.1)	339 to 1014	+1431 to 4293 I
1.2.1.2.1	Crew Quarters Module	o	Revise envelope to 10mØx14m long	+494	+2528 I
		o	Revise interior arrangement		
		o	Revise quantity of crew quarters to reflect both the smaller crew size/module and the increased number of crew members.		
1.2.1.2.2	Work Modules	o	Make same revisions as described for WBS 1.2.1.2.1	+393	+1397 I
1.2.1.3	Operations	o	Add 56 crew members to crew size (additional crew for additional solar array deployment and subassembly operations).		
			o Crew habitat operations crew (+28)	(+28)	+9.8 Ø
			o Solar array crew (+8)	(+8)	+1.03 Ø
			o Slip ring subass'y crew (+8)	(+8)	+1.03 Ø
			o Cargo pallet ops crew (+12)	(+12)	+2.6 Ø
		o	Add additional supplies for additional crew size, more crew modules, additional cherry-pickers, etc.		+71.4 Ø

[1] Cost Category Code: P = Production; I = Investment; Ø = Operations

[2] Cost of transporting additional mass has not been included.

TABLE 2.3-2 (Con't)

EFFECTS OF THE SMALLER HLLV ON THE SPS PROGRAM

WBS	ITEM		DESCRIPTION OF EFFECT	MASS MT	COST $M [1] [2]
	Leo Base				
1.2.2.1.2	Construction Equipment	o	Add one set of solar array deployment equipment	+12 I	+45 I
1.2.2.1.3	Cargo Handling/Dist	o	Due to smaller and more numerous cargo pallets:	0	0
			o Cargo Pallet Handling Machine—1 more req'd—revise to smaller size	0	0
			o 20 cherrypicker—18 more req'd for cargo sorting	135	+394 I
			o Pallet Handling Jig—2 more req'd —revise to smaller size	1	+1.8 I
			o Cargo Transporters—60 more req'd	30	+90 I
			o Cargo Sorting Systems—add 9 units	22.5	+207.9 I
			o Crew Transfer Tunnel Systems—add 3 systems	6	+3 I
1.2.2.1.4	Subassembly Factories	o	Revise the thruster subassembly factory for smaller thruster panel subassemblies.	0	0

[1] Cost Category Code: P = Production; I = Investment; Ø = Operations

[2] Cost of transporting additional mass has not been included.

TABLE 2.3-2 (Con't)

EFFECTS OF THE SMALLER HLLV ON THE SPS PROGRAM

WBS	ITEM	DESCRIPTION OF EFFECT	MASS MT	COST $M [1] [2]
1.2.2.1.6	Space Transportation Support Systems	o Add 3 HLLV docking systems o Smaller size o 2 dedicated to propellant tankers	7.2	+16.6 I
		o Add 2 cargo tug docking systems	1	+30 I
		o Add propellant transfer, storage, and conditioning system (assume EOTV propellant pallets can be assembled at LEO Base)	35	+15 I
1.2.2.2	Crew Support System	o Revise crew and work habitat modules per WBS 1.2.1.1	+596	+3026 I
1.2.2.3	Operations	o Revise supplies list (space parts) to reflect changes in crew modules & subsystems.		+45.5 Ø
		o Revise crew salaries		+9.5 Ø
1.2.2.3.1	EOTV Construction Operations	o Revise solar array deployment ops to account for addition deployment system.		[3]
		o Add 4 crew members for additional thruster subass'y		[3]
		o Add 4 crew members for other subassembly		[3]

[1] Cost Category Code: P = Production; I = Investment; Ø = Operations

[2] Cost of transporting additional mass has not been included.

[3] Crew costs accounted for under WBS 1.2.2.3

TABLE 2.3-2 (Con't)

EFFECTS OF THE SMALLER HLLV ON THE SPS PROGRAM

WBS	ITEM		DESCRIPTION OF EFFECT	MASS MT	COST [1] $M [2]
	MOBILE MAINTENANCE				
1.2.2.3.2	Logistics Operations	o	Revise the HLLV and EOTV operations to reflect more numerous operations/day		
			o Cargo pallet handling ops—add 20 people		[3]
			o Docking propellant handling ops—add 8 people		[3]
1.2.3	Mobile Maint. Support Systems	o	Revise to reflect changes in crew habitat size (see WBS 1.2.3.2) and OTV resizing (see WBS 1.3.4)		
1.2.3.2	Crew Support System	o	Revise crew habitat module per WBS 1.2.1.2		+2 I
1.3	**SPACE TRANSPORTATION**	o	Revise transportation scenario to reflect:		
			o More HLLV flights w/reduced payloads		
			o Revised resupply mass (as modified for new crew modules, additional people, etc.)		
			o More cargo tug operations		
			o Revised POTV operations—trips		

[1] Cost Category Code: P = Production; I = Investment; Ø = Operations

[2] Cost of transporting additional mass has not been included.

[3] Crew costs accounted for under WBS 1.2.2.3

TABLE 2.3-2 (Con't)

EFFECTS OF THE SMALLER HLLV ON THE SPS PROGRAM

WBS	ITEM	DESCRIPTION OF EFFECT	MASS MT	COST [1] $M
1.3.1	HLLV	o Total revision		
1.3.2	EOTV	o Revise cargo platform for more and smaller cargo pallets	+1	+1 I
		o Revise thruster panel configuration to show 4 sub-panels		
1.3.4	POTV	o Modify OTV to fit within HLLV		
1.3.5	Orbital Personnel Module	o Modify OPM to fit within smaller HLLV		
1.3.6	Cargo Tug	o Add 4 cargo tugs (2@ LEO, 2@ GEO)	+40	+100 I
1.3.7	**GROUND SUPPORT FACILITIES**	o Reference location (Kennedy Space Center) may have to to be changed due to more frequent HLLV operations		
1.3.7.1.1	HLLV Launch Facilities	o Add 3 more launch systems—smaller size		-2273 I

[1] Cost Category Code
P = Production
I = Investment
Ø = Operations

TABLE 2.3-2 (Con't)

EFFECTS OF THE SMALLER HLLV ON THE SPS PROGRAM

WBS	ITEM		DESCRIPTION OF EFFECT	MASS MT	COST [1] $M
1.3.7.2.2	HLLV Orbiter and Payload Processing Facility	o	Revise size of bays and quantity of bays req'd to accommodate smaller and more numerous orbiter stages.		-849 I
1.3.7.2.3	HLLV Booster Processing Facility	o	Revise size of bays and quantity of bays req'd to accommodate smaller and more numerous booster stages.		-244 I
1.3.7.2.4	Engine Maintenance Facility	o	Revise size and support equipment to accomodate different size engines and larger quantities		0
1.3.7.3	Fuel Facilities	o	Revise as req'd to reflect possible new launch site location and the more frequent HLLV launch ops.		0
1.3.7.5	Operations Facilities	o	Revise to reflect more frequent HLLV operations		+78I
1.3.7.6	Operations	o	Revise headcount to reflect more frequent HLLV ops.		+239 Ø
1.5	**OPERATIONS CONTROL**				
1.5.1	Facilities and Equip.	o	Revise to reflect more people associated with HLLV operations and maintenance		+189.7 I
1.5.3	Operations	o	Revise headcount to reflect more people associated with HLLV operations and maintenance. = 1084 people @ $50k/yr		+542 Ø

[1] Cost Category Code: P = Production; I = Investment; Ø = Operations

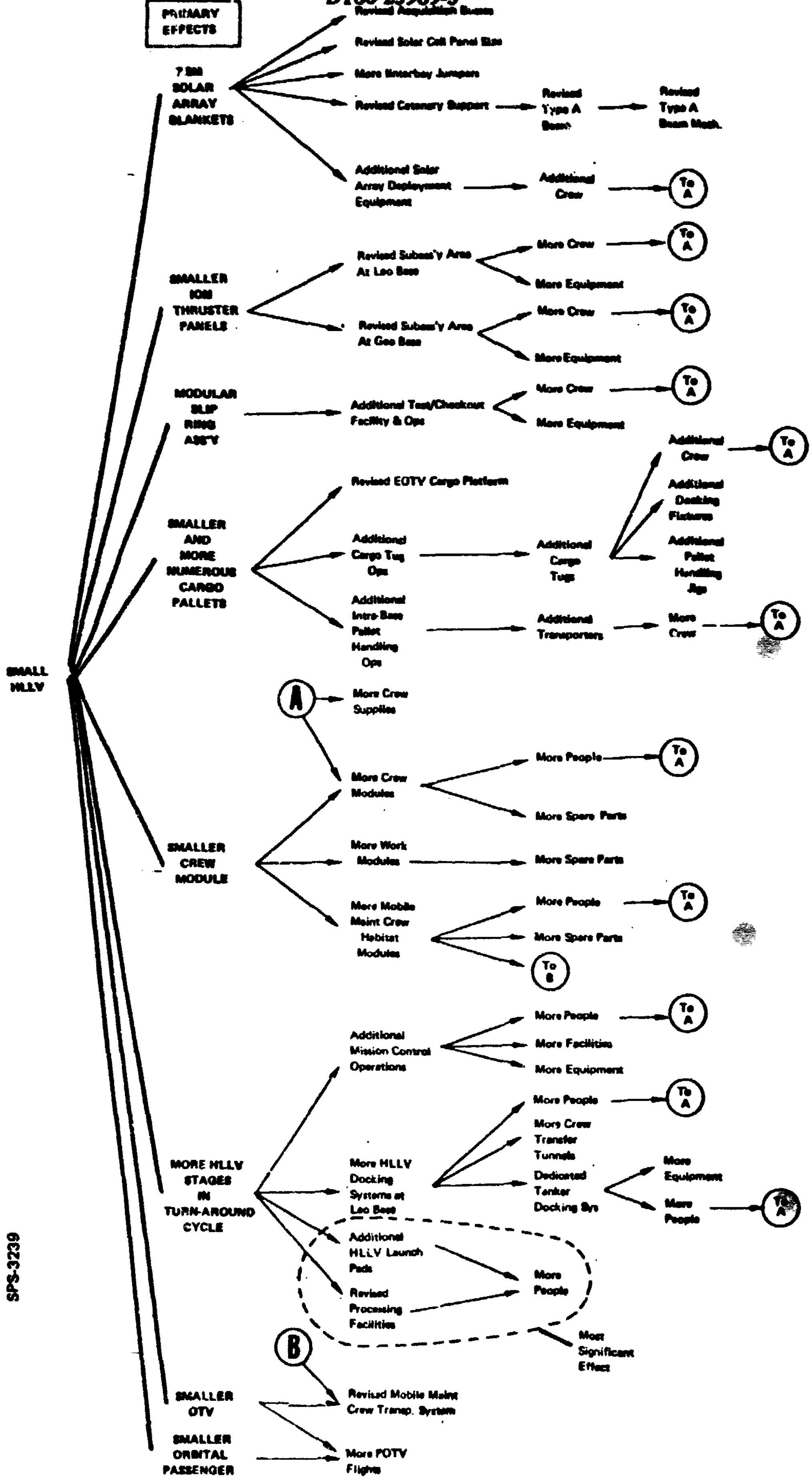


Figure 2.3-1. Interrelationships of the Effects of a Small HLLV

TABLE 2.3-3
ANALYSIS OF PRIMARY EFFECTS

EFFECT	ANALYSIS
o 7.5m Solar Array Blankets	o Anything less than 15m leads to problems. o [illegible] cargo bay could be in excess of 15m long and if the blankets could be shipped on end, then there would be no impact.
o Smaller Ion Thruster Panels	o The thruster panels were to be assembled from 2 subassemblies anyway, so having to assemble from 4 subassemblies is of only minor impact.
o Modular Slip Ring Assy's	o Anything less than 16m diameter is a problem. o The assembly could be knocked down into cylindrical quadrants.
o Smaller and More Numerous Cargo Pallets	o Smaller size units offset some of cost associated with having more units. o There is some quantity of additional units that could be tolerated before exceeding the capabilities of the presently defined set of handling equipment and crew.

TABLE 2.3-3 (Cont'd)
ANALYSIS OF PRIMARY EFFECTS

EFFECT	ANALYSIS
o Smaller Crew Modules	o The smaller HLLV leads to a 20 man crew habitat, see Section 2.3.2.1.2.
o More HLLV's	o With only 3 launch pads and a 7-day/2-shift launch schedule, only 1 or 2 more launches per week could be realistically scheduled. o Each launch pad can support only 2.5 launches per week (on a 7-day/2-shift schedule). o Going to a 3 shift schedule, 7 days per week, each launch pad can support 3.75 launches/week. o 6 pads will be required. (2 alternative arrangements of 6 HLLV launch pads at KSC are described in Section 2.3.2.1.3) o A 7-day/week, 24 hr/day launch schedule will probably be environmentally unacceptable (noise level). Therefore, a remote, equatorial launch site would probably be required. o The largest cost associated with launch pads is the taxiways and offshore causeways and break waters (over 70% of cost). o The LEO Base will have to have at <u>least</u> 3 additional HLLV Docking Systems.

TABLE 2.3-3 (Cont'd)
ANALYSIS OF PRIMARY EFFECTS

EFFECT	ANALYSIS
o Smaller OTV	o Redesign OTV to be shorter and larger diameter and still keep baseline performance capability see Figure 2.3-2.
o Smaller Orbital Passenger Module	o Could redesign to a shorter, larger diameter stage with double deck to keep 75 passenger capacity, see Figures 2.3-3 and -4.

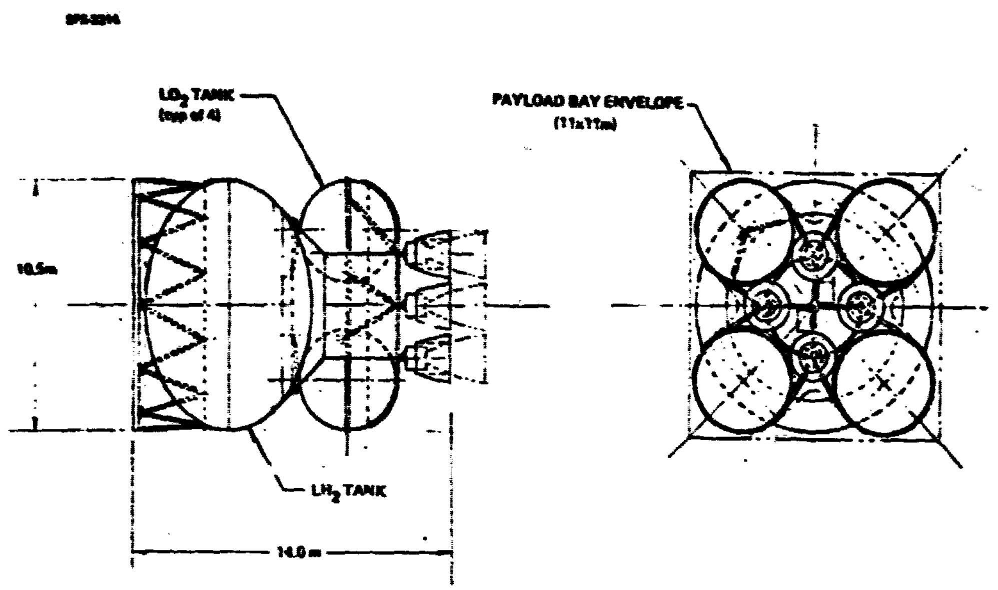
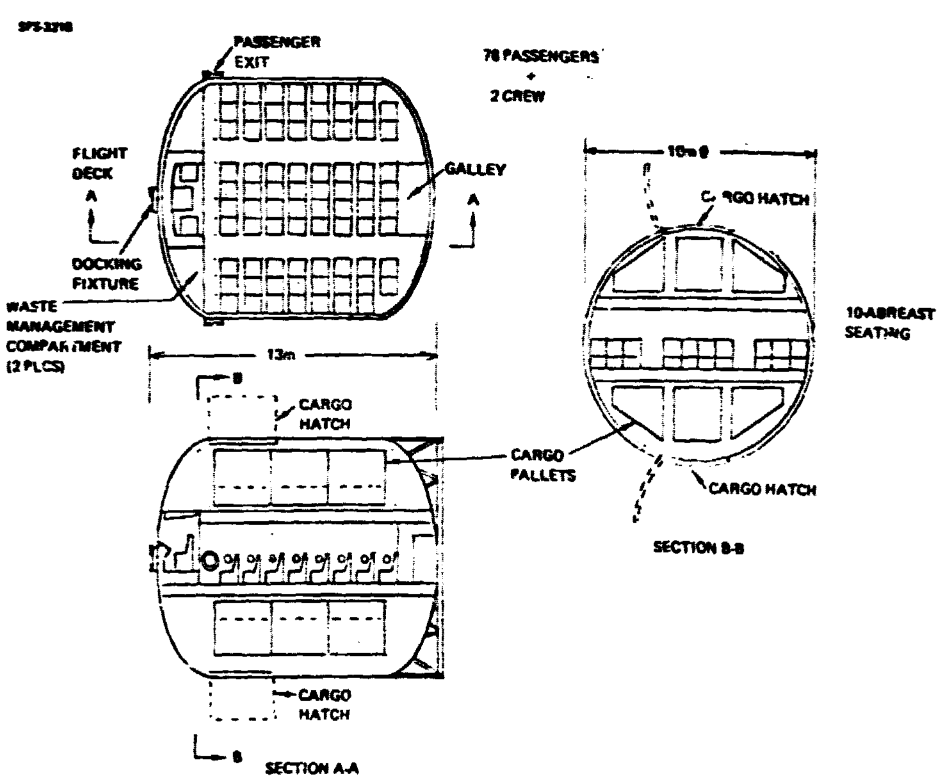


Figure 2.3-2. Orbital Transfer Vehicle Configured to fit within a Small HLLV



. .gure 2.3-3. Orbital Passenger Module Configured to fit within a Small HL.LV

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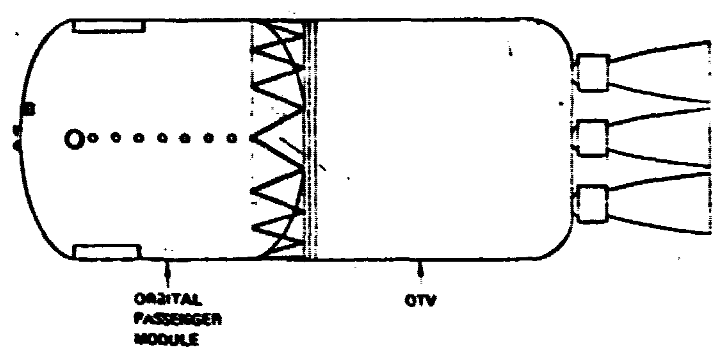


Figure 2.3-4. Personnel Orbit Transfer Vehicle (POTV)

examined to find the components that 1) would be affected by the smaller cargo bay envelope, and 2) those that are either the most numerous, the most massive, and/or the largest (the so-called "primary payloads"). These components are identified in Figure 2.3-5.

When comparing the small HLLV "primary payloads" identified in Figure 2.3-5 against the "primary payloads" identified in Figure 5-5 of the Reference, it will be noted that the Antenna Secondary Structure and the Propellant Pallets have not been included in Figure 2.3-5.

The Secondary Structure package has changed for the baseline system (since the Reference was published) to a fabricated structure instead of a deployable structure. The material for this fabricated structure will be beam machine roll stock and has, therefore, been included into the combined beam machine feed stock shown in Figure 2.3-5.

The POTV, SPS, and EOTV Propellant Pallets have been deleted as it is assumed that there will have to be dedicated HLLV tankers.

The only components that are repackaged significantly are the solar array blankets, the ion thruster panels, and the electrical rotary joint (slip ring) assembly.

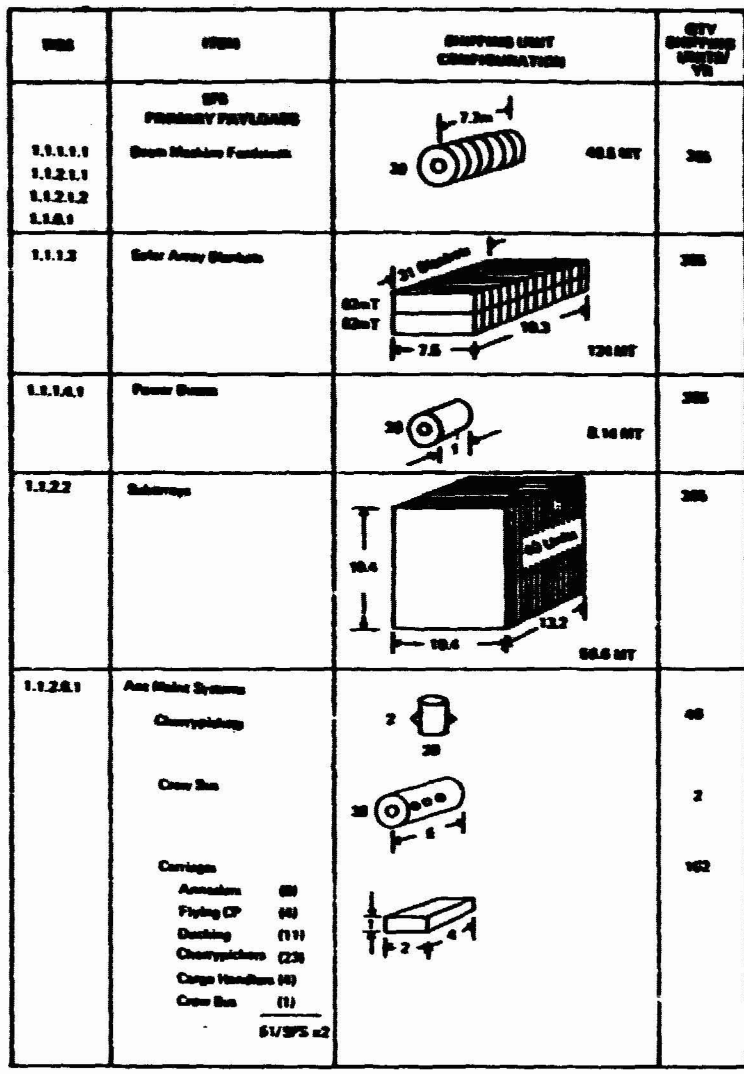


Figure 2.3-5. Primary Payloads

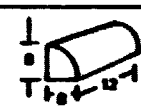
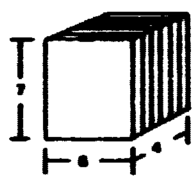
WBS	ITEM	SHIPPING UNIT CONFIGURATION	QTY SHIPPING UNITS/ YR
[illegible]	Slip Ring Ass'y	 [illegible] MT	[illegible]
1.1.4	Thruster Panels	 [illegible] MT	1
1.2.2.1.1	[illegible] PRIMARY PAYLOADS [illegible] Feed Stock	 4.8 MT	120
1.2.2.3	Solar Array Blankets	 10.3 MT	345
1.2.2.4	Thruster Panels	 10 MT	8
	Crew Supply Modules		40 (YR1) to 160 (YR30)

Figure 2.3-5. Primary Payloads (Continued)

2.3.2.1.2 GEO BASE IMPACTS FROM SMALLER HLLV

Smaller payload capability of the HLLV reduces the allowable cargo size and mass that can be delivered into low earth orbit. At the GEO construction base, however, the reduction in HLLV payload size will be important. The 11m x 11 m x 14m cargo bay limitation lead to alternate SPS construction requirements, which impact GEO base systems as shown in Fig. 2.3-6. When more construction tasks are added, extra equipment and/or work areas are needed. The smaller cargo bay also limits the size and hence the number of required pressure vessels for habitation and work support functions. A greater number of small cargo containers must be handled and distributed through the intra base logistic network. All of the above leads to a larter crew, additional housing, more base support structure, etc.

Figure 2.3-7 shows the Phase 2 reference construction base and the alternate base which relies on the smaller HLLV. The alternate base, which uses smaller crew modules, is 14% heavier and requires a larger crew to maintain the reference production rate. Although the alternate GEO base has a higher unit cost, it also shares a lower development cost with the LEO base crew module. The smaller crew module provides a significant reduction in DDT&E expenditures at the outset of the investment phase. As a result, the initial investment costs (DDT&E & unit) will only be 50% greater than the reference base. The full deployment cost of the crew module could also be deleted from the investment phase if the smaller module was developed for common use by the preceeding SPS demonstration phase. The following paragraphs discuss the major effect of the smaller HLLV on GEO base operations and related crew support facilities.

GEO Base Operations Impact - The smaller HLLV cargo bay (11 m x 11 m x 14 m) affects GEO base operations for satellite construction and intra base logistics. In particular, increased construction requirements lead to additional equipment and crew staffing for the intra base logistics system as well as for construction.

Revised satellite construction requirements include smaller solar array blanket cannisters (7.5 m vs 15 m), modifications to solar blanket interfaces (e.g., support structure, acquisition buses, etc), and modular versus preassembled slip rings. Those operations, which impose added equipments for the GEO base, are listed in Fig. 2.3-8 with their system impacts (i.e., delta mass and cost). To maintain the six month reference construction schedule, twice as many cherry pickers are needed to install 88 versus 44 solar array blankets in each bay of the energy conversion system. No additional equipment is needed to handle the other subsystems which interface with the smaller solar array blankets. However, the Level J subassembly factory must be

- SMALLER HLLV PAYLOAD CAPABILITY
  - 11 X 11 X 14 m VS 17 m DIA X 23 m CARGO BAY
  - 120 MT VS 400 MT
- ALTERNATE SPS CONSTRUCTION REQUIREMENTS
  - 7.5 m VS 15 m SOLAR ARRAY BLANKETS
  - MODULAR VS ASSEMBLED SLIP RING DELIVERY
- GEO BASE SYSTEMS IMPACT
  - ADDED EQUIPMENT/WORK AREAS
  - SMALLER HABITATS & WORK MODULES
  - MORE INTRA-BASE LOGISTICS
  - LARGER WORK FORCE
  - ADDITIONAL BASE STRUCTURE

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Fig. 2.3-6 Smaller HLLV Payload Effects on GEO Construction Base

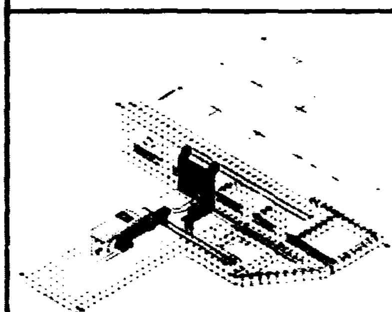
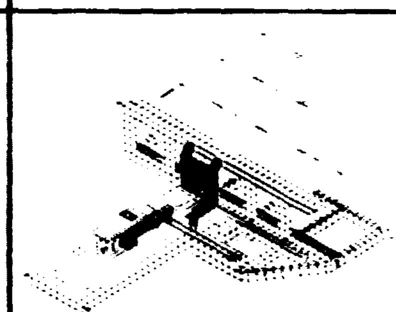
	BASELINE	BASELINE WITH SMALLER HLLV
		
• SPS PRODUCTION RATE	10 GW/YR	10 GW/YR
• CREW MODULE DEVEL COST, 1979$	$ 5.16B	$ 3.78B
• BASE UNIT COST	$ 9.01B	$15.17B
• BASE ANNUAL COST	$ 1.30B/YR	1.46B/YR
• BASE MASS	6656 MT	7707 MT
• GEO CONSTRUCTION CREW	444	500
0847-002W		

Fig. 2.3-7 Alternate SPS Construction Bases

expanded to accommodate the equipment needed to support the assembly and checkout of the modularized slip ring. Finally, it is estimated that four times as many cargo pallets must be docked/unloaded and handled.

GEO base crew operations are also increased to support the added tasks for satellite construction and intra base logistics. It is estimated that 56 crewmen will be needed to cover the extra workload and furnish the required habitat and crew support services. Figure 2.3-9 shows a breakdown of theses added crew operations, together with the extra cost for annual operations.

Crew Support Facilities Impact - The reduced size cargo bay of the small HLLV results in a smaller pressurized module to support habitation and work-related activities. This module is now 10.5 m dia. x 13.5 m instead of the 17 m dia. x 23 m long module that the reference HLLV can transport. Figure 2.3-10 considers the number of small modules necessary to replace one large module.

In the Phase 2 analysis of crew habitation requirements, it was judged that one large module, sized for the reference HLLV could comfortably house 100 men. On a direct volume basis, five of the smaller modules would provide approximately the same volume as one larger module. (In fact, the equivalent volume ratio is probably greater than 5 to 1, since packaging given items into a smaller volume is less efficient than packaging the same items into a larger volume. This holds for all crew support facilities where the initial allocation of functional areas is either believed to be correct or is perhaps not well defined.) The GEO base work modules for command and control, base maintenance, etc have yet to be analyzed. When the functional requirements for these activites are developed, the area needed for crew and equipment could either meet or exceed the current assumptions. Hence the 5 to 1 ratio is used to establish equivalent work modules for the smaller HLLV. Crew habitation requirements, however, were examined in Phase 2 to the level of compartmental partitioning of major crew areas, considering furnishings and equipment. The larger crew module provided about 17.44 m^3 of free volume for each crewman. This is about 2.5 times Celentano's recommended free volume per man (7.08 m^3) for acceptable crew performance over 90 days. Therefore, a brief study was performed to take another look at the crew accommodation packaging arrangements for the smaller crew module. By reducing the free volume crew allocation to 10.35 m^3, we judge that 100 men can be adequately housed in three of the smaller modules.

REVISED OPERATIONS	GEO BASE SYSTEM IMPACT		
	ADDED EQUIPMENT	Δ MASS	Δ COST
• INSTALL 88 - 7.5 m SOLAR ARRAY BLANKETS/BAY (TWICE BASELINE)	(4) 30 m CHERRY PICKERS @ LEVEL H ANCHORS	10 MT	$ 87.6M
• ASSEMBLE & C/O MODULAR SLIP RING	(2) 30 m CHERRY PICKERS, RACKS & TOOLS, TEST & C/O EQUIP. @ LEVEL J FACTORY	15 MT	$ 97.6M
• DOCK/UNLOAD & HANDLE MORE NUMEROUS SMALL CARGO PALLETS (FOUR TIMES BASELINE)	(2) CARGO TUG DOCKING PORTS (2) CARGO PALLET HANDLING JIG (80) TRANSPORTERS (SMALL) @ LEVEL J	22 MT	$ 78.8M
0847-003W		47 MT	$264M

Fig. 2.3-8 GEO Construction Operations Impact Due to Smaller HLLV

			CREWMEN
• BASELINE GEO CONSTRUCTION CREW			444
• ADDED CREW OPERATIONS			56
	– SOLAR ARRAY INSTALLATION	8	
	– SLIP RING ASSEMBLY & C/O	8	
	– CARGO HANDLING & DISTRIBUTION	12	
	– HABITAT & CREW SUPPORT (UTILITIES, HOTEL, FOOD MGT, MAINT, ETC)	28	
	ADJUSTED BASE CREW		500
• OPERATIONS COST IMPACT			
	– ADDED CREW SALARIES		$83.3M
	– ADDED CREW SUPPLIES ($1.43M/MANYR)		$80.1
0847-004W			$163.4M

Fig. 2.3-9 Effect of Smaller HLLV

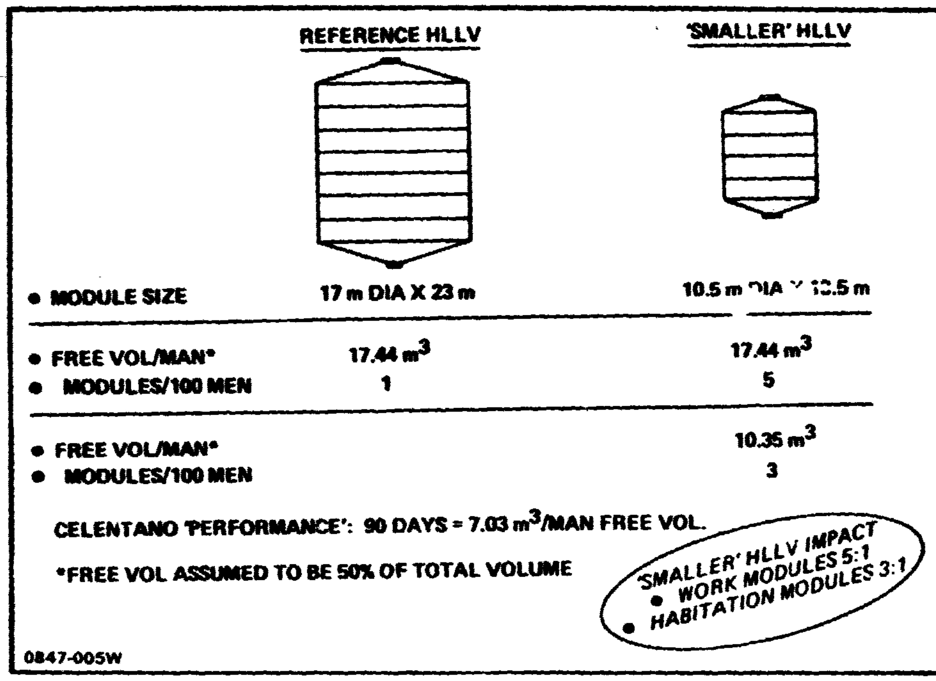


Fig. 2.3-10 Impact of HLLV Size on GEO Base Modules

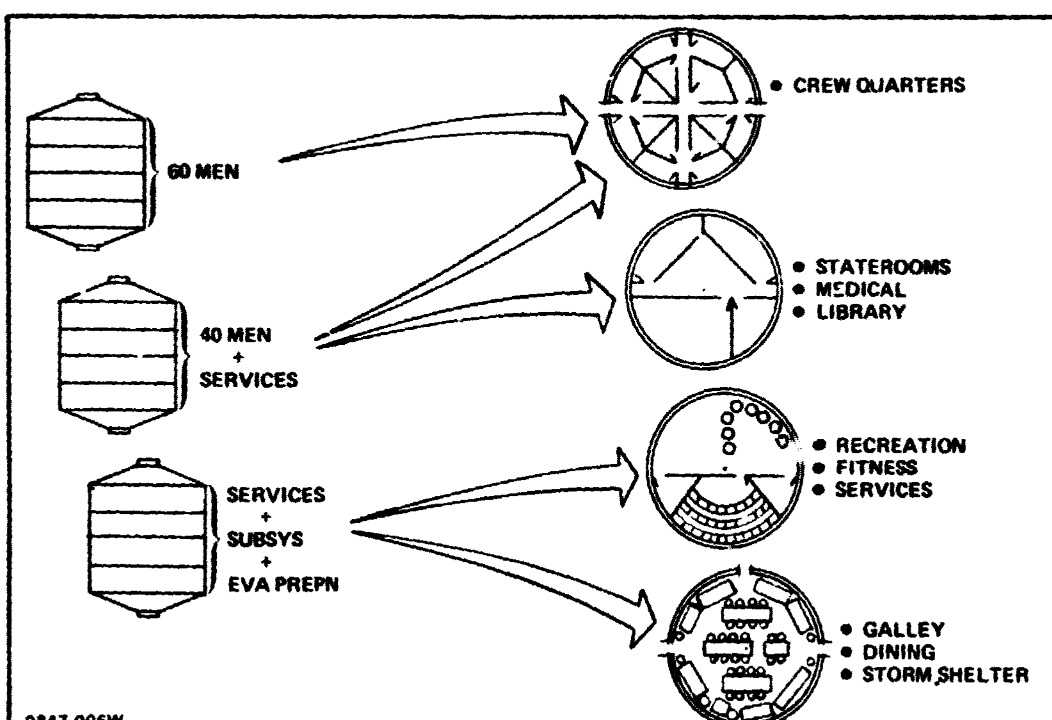


Fig. 2.3-11 Crew Module Size for "Smaller" HLLV Launch – Three Modules House 100 Men

Revised layouts for these smaller habitation modules are shown in Figure 2.3-11. Allowing for wall thickness, insulation and radiation protection, the inside diameter of each deck is 10 m and floor to ceiling height is 2.15 m. One module provides quaters for 60 men and each of the four decks has the same layout of 16 comparably sized quaters; except that on two of the decks, two quarters are eliminated on each to provide hygiene and waste management. The second module has one deck of 14 quarters plus toilets, laid out as the first module, then two decks with 12 larger quarters each. A fourth deck provides medical facilities, a library and two staterooms for the two most senior officers. The third module provides services on two of the four decks. One deck provides a gymnasium, a recreation lounge, a thirty seat theatre for movies, church services and meetings, a laundry and a hygiene/waste management facility. The other service deck has the galley, food storage for emergencies and eating accommodation for 28. Main food stroage is in an attached logistics module. This deck also serves as the storm shelter with suitable distribution of equipments and wall thicknesses to provide protection. The free area available for 100 men during solar storm events is $0.54 m^2$ ($5.8 ft^2$) per man. The remaining two decks in this module house subsystems and EVA preparation.

Comparison of the smaller module to the larger baseline module, Fig. 2.3-12 shows, as alternates, the estimated total number of GEO base crew support facilities. Mass and cost data are shown for each module and the estimated penalty is identified for the smaller module. The number of crew habitats and related work modules are defined for support of GEO construction and SPS maintenance. When the appropriate small module to baseline module ratio is applied (i.e., 3:1 habitats and 5:1 work), 33 small modules (10.5 m dia) are required for initial GEO construction (vs 8 at 17 m dia). Later in the program when 60 satellites have to be maintained, 99 of the smaller modules will be needed for habitation and work support functions.

Figure 2.3-13 shows a comparative breakdown of the major elements covered by the estimates for crew module mass and average unit cost. The smaller module retains the reference cabin wall design for protection against trapped electron flux. A one deck storm shelter is also provided, as in the reference, for environmental protection against solar flares. Environmental control subsystem weights are based on 60 men, as defined in Fig. 2.3-11. Weight estimates for the other subsystems of the small module (i.e., communications, electrical power and crew accommodations) are also adjusted for the 60 man crew. As shown in Fig. 2.3-13, the latter subsystems

	BASELINE 17 m DIA X 23 m	SMALLER HLLV PAYLOAD 10.5 m DIA X 13.5 m	Δ MASS, MT	Δ COST, $
GEO CONSTRUCTION SUPPORT				
• CREW HABITATS	5	18		
– TOTAL (UNIT) MASS, MT	1215 (243)	1710 (95)	494	
– TOTAL (AVG UNIT) COST, 1979 $M	1923 (384.6)	4451 (247.3)		2528*
• WORK MODULES	3	15		
– TOTAL MASS, MT	413	807	393	
– TOTAL COST, $M	631	2028		1397*
SPS MAINTENANCE SUPPORT (20 TO 60 SATELLITES)				
• CREW HABITATS	4 TO 12	12 TO 36		
– TOTAL MASS, MT	972 – 2916	1140 – 3420	168 TO 504	
– TOTAL COST, $M	1538 – 4615	2967 – 8903		1429–4288*
• WORK MODULES	2 TO 6	10 TO 30		
– TOTAL MASS, MT	354 – 1062	692 – 2076	339 TO 1014	
– TOTAL COST, $M	646 – 1938	2077 – 6231		1431–4293*
		TOTAL Δ MASS	1393 TO 2405 MT	
*EXCLUDES FULL BENEFITS OF LEARNING		TOTAL Δ COST		$ 6,785M TO $12,500M*

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Fig. 2.3-12 HLLV Impact on GEO Base Crew Support Facilities

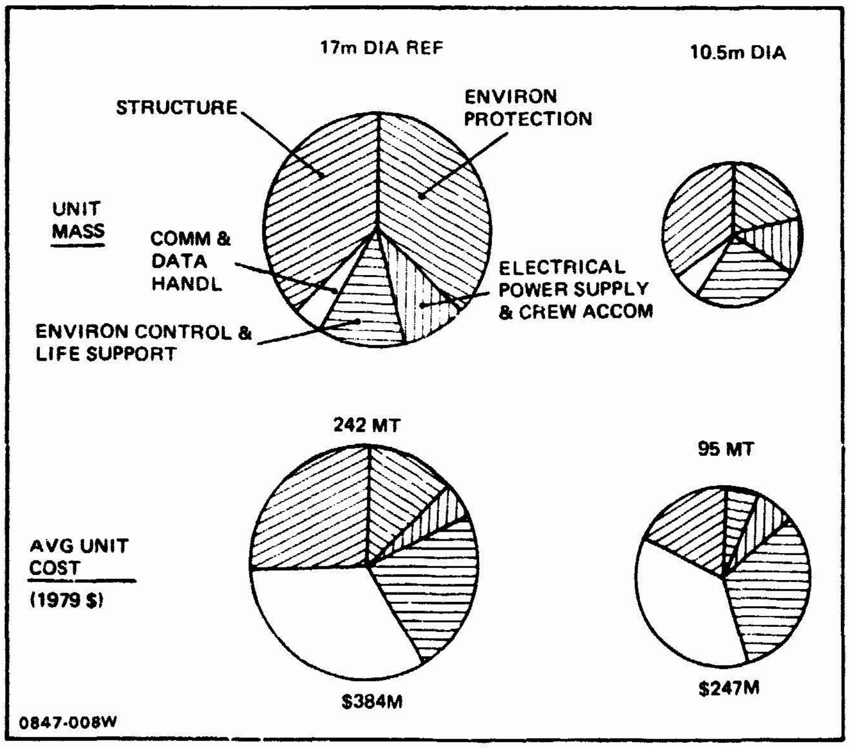

Fig. 2.3-13 Crew Module Comparison – Mass & Cost

represent less than 25% of the reference module mass but almost half of the smaller module mass. From a cost point of view, the latter subsystems account for more than half the cost of either module. This is because these subsystems contain basic components (fixed costs) which are insensitive to changes in crew size or module geometry. Lower crew module costs are possible, of course, if the smaller modules were defined differently and compared in terms of their respective functions and capabilities. It should be noted that the cost penalty attributed to the smaller pressure vessel in Fig. 2.3-12 is probably too high since these cost data do not include the full benefit of production quantity learning.

The large number of crew modules resulting from the smaller HLLV raises the question as to how they might be accommodated on the base. The center of GEO base logistic activities occurs at the top deck, Level J, which includes the crew quarters/operations center and areas for growth. For example, at the end of the 30 year reference scenario, the crew quarters/operations complex could grow to 99 modules. Figure 2.3-14 shows that Level J has ample area to mount as many small modules as needed.

Net Impact of Smaller HLLV on GEO Base - The net impact of the smaller HLLV on GEO base mass and cost is summarized in Fig. 2.3-15. The reference work facilities must be revised primarily to support the added crew support facilities, accommodate extra construction equipment, enlarge cargo handling/distribution, and expand the subassembly factory. One benefit of the smaller crew module is that it provides a significant reduction in DDT&E expenditures which occur at the outset of the investment phase. It also provides a programmatic option that would make one crew module size serve needs for both the demonstration and investment phases of the program. In that event, only one module would be developed and funded to meet earlier demonstration phase objectives. This option would then avoid $3.8B (with wraparound factors) for developing another small crew module for the investment phase.

It should be noted again that the crew module production costs are probably too low since they exclude the full benefits of high production learning. In addition, the range of crew modules costs cover an expenditure over 30 years with no discounting included.

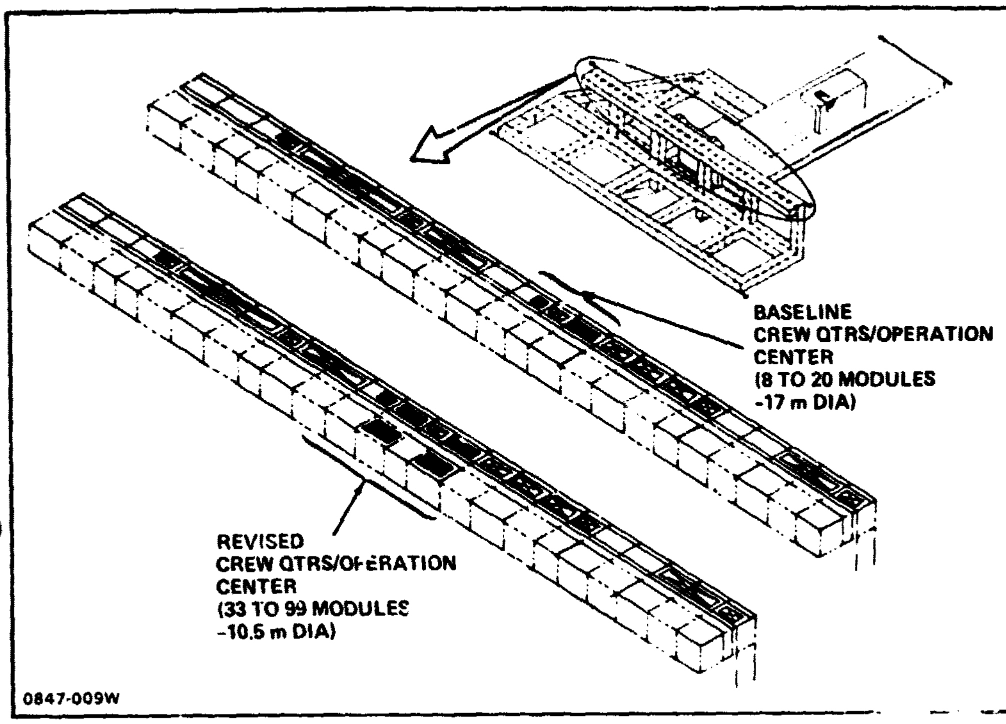

Fig. 2.3-14 GEO Base: Level J Facilities – Impact of Smaller HLLV

GEO BASE ELEMENT	Δ MASS, MT	Δ COST 1979 $M DDT&E	Δ COST 1979 $M PROD.
WORK FACILITIES			
– STRUCTURE	17	4	2
– CONSTRUCTION, EQUIPMENT	10	0	88
– CARGO HDLG/DISTRIBUTION	22	0	79
– SUBASSEMBLY FACTORIES	15	0	97
CREW SUPPORT FACILITIES			
– CREW QUARTERS (0 TO 60 SPS)	494 TO 998	-613	2528 TO 6816
– WORK MODULES	393 TO 1407	0	1397 TO 5690
WRAPAROUND FACTORS			
– DEVMT 127%		-773	
– PROD. 47%			1969 TO 6002
TOTAL	951 TO 2469 MT	-$1380M	$6160 TO 18770M
		$4,780 TO $ 17,390M	
ANNUAL OPERATIONS			
SALARIES & TRAINING (+56 CREW)			83
RESUPPLY	142 MT/YR		80
			$163M/YR

SMALL HABITAT OPTION
• KEY TO DEMO PHASE
• AVOID $3.8B INVEST.

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Fig. 2.3-15 Net Impact of Smaller HLLV on GEO Base

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2.3.2.1.3 Alternative Launch and Recovery Site Concepts

In the analysis of the effects of a small HLLV on the SPS program elements, it was found that one of the most significant effects would be on the launch and recovery site. This analysis was prepared to amplify the basis of this assessment and to show some alternative solutions.

Calculation of the Number of Launch Pads—In Table 2.3-1, it was shown that at year 12 (when 20 SPS's are in orbit, per year) that 1471 mass-limited flights would be required. Multiply this by 1.05 to account for non-optimal packaging and we get 1545 flights per year. The pad time per vehicle is 34 hours. This leads to the capability of each pad to support 257 flights per year (assuming 24 hours per day/365 days per year operations). This results in a requirement for 6 launch pads for the small HLLV.

Launch Pad Locations—If we assume that it will be environmentally acceptable to launch up to 5 vehicles per day every day of the week at KSC, then we are given the requirement to find space for 6 HLLV launch pads. In Task 4210111, we found that for the small HLLV that the minimum pad separation distance required is 8000 ft.

We examined 2 possible arrangements of 6 HLLV launch pads at KSC that meet the 8000 ft separation requirement. Figure 2.3-16 shows an off-shore arrangement similar to the baseline concept for the large HLLV. Figure 2.3-17 shows an arrangement where the 6 pads are located on-shore. In this arrangement, 3 of the HLLV pads will be at the 38C, 39D, and 39E pad locations (shown to be in locations previously reserved for them). The 3 additional HLLV pads are shown to be located at the 37, 40, and 41 pad locations. (It is assumed that the current user of these pads will no longer be operational or that they can be moved to other pad locations. In addition, pads 34, 20, and 19 will have to be demolished to provide the 8000 ft clearance).

Cost Analysis Highlights—The cost estimates for the alternative launch and recovery sites are summarized in Table 2.3-4. The 5 alternative concepts are described below:

- **Large HLLV—Reference**
  - This is the reference concept for the large HLLV, described in the Reference System Description, WBS 1.3.7.
- **Large HLLV—Piers**
  - This concept substitutes a 200 ft wide steel pier system in lieu of the rock causeways. Brown and Root estimates this steel pier arrangement to cost $50,000 per lineal foot.
- **Small HLLV Causeways**
  - This arrangement of this concept is shown in Figure 2.3-16.
  - The causeways are 100 ft wide and 50 ft high.
  - The launch pads are scaled to be 35% as large and expensive as that required for the large HLLV.
  - The HLLV Orbiter and Booster processing facilities were scaled down to the smaller vehicle sizes and additional bays were provided as required. Scaling down the vertical clearance height and the strength required resulted in substantial cost savings.

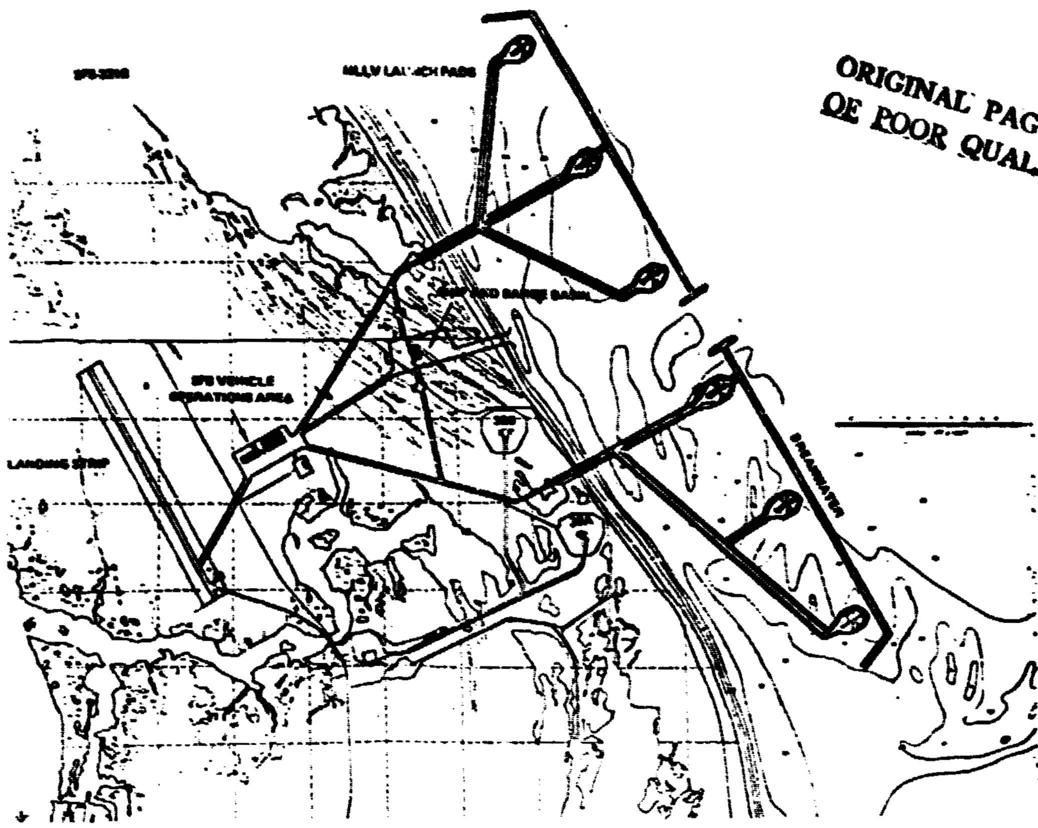

Figure 2.3-16. SPS Launch and Recovery Site Arrangement at KSC Configured for a Small HLLV

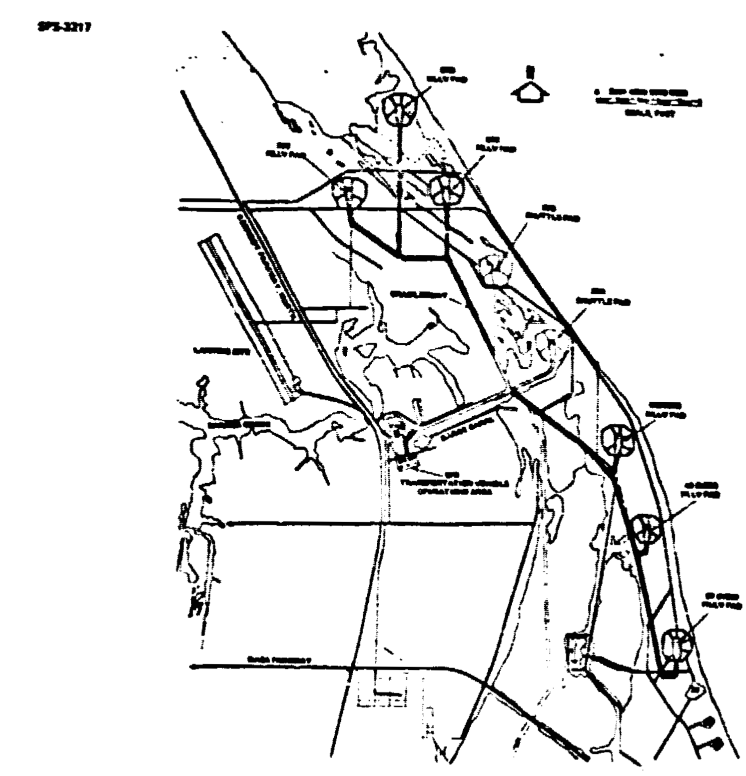

Figure 2.3-17. On-Shore Arrangement of SPS Launch and Recovery Site Facilities at KSC Configured for a Small HLLV

TABLE 2.3-4

COST COMPARISON OF ALTERNATIVE LAUNCH AND RECOVERY SITE CONCEPTS

		COST, $M				
		LARGE HLLV		SMALL HLLV		
WBS	**ELEMENT**	**REFERENCE**	**PIERS**	**CAUSEWAY**	**PIERS**	**SHORE**
1.3.7.1.1	HLLV Launch Facilities	(3222)	(3345)	(3828)	(4828)	(949)
	o Causeways & Taxiways	1727	1850	1950	2950	180
	o Breakwater	673	673	1109	1109	–
	o Launch Pads	336	336	234	234	234
	o Equip/utilities/etc.	486	486	535	535	535
1.3.7.2	Recovery Facilities	(1770)	(1770)	(676.5)	(676.5)	(676.5)
1.3.7.2.1	o Landing Site	20.5	20.5	20.5	20.5	20.5
1.3.7.2.2	o HLLV Orbiter Proc. Fac.	1114	1114	265	265	265
1.3.7.2.3	o HLLV Booster Proc. Fac.	445	445	201	201	201
1.3.7.2.4	o other facilities	190	190	190	190	190
1.3.7.2.10						
1.3.7.3	Fuel Facilities	TBD	TBD	TBD	TBD	TBD
1.3.7.4	Logistics Support	(40)	(40)	(40)	(40)	(6)
1.3.7.5	Operations	(78.3)	(78.3)	(156.6)	(156.6)	(156.6)
	INVEST. TOTALS	$5.11B	$5.23B	$ 4.7B	$ 5.7B	$1.8B

- **Small HLLV Piers**
  - This arrangement for this concept was identical to that described above.
  - The only difference is that 100 ft wide steel piers are used in lieu of the rock causeways. Brown and Root estimated the cost to be $42,000 per lineal foot.

- **Small HLLV On-Shore**
  - The arrangement for this concept was shown in Figure 2.3-17.
  - The ship and barge basin were eliminated.
  - The scaled-down orbiter and booster processing facilities were also used here.
  - The cost of the new causeway was included.

RECOMMENDATIONS—It is obvious that the so-called "on-shore" pad arrangement is substantially cheaper than the "off-shore" alternatives. These cost estimates were fairly crude, so it is suggested that a task be provided in future studies to derive more detailed cost data.

The environmental effects of a 24 hour per day, 7 day per week launch schedule cannot be ignored. A more detailed study is required to define the maximum launch rate that could be tolerated at KSC.

2.3.3 Conclusions

The mass and cost deltas associated with each of the 8 primary effect chains are summarized in Table 2.3-5. It is evident that the smaller crew modules are the dominating effect.

TABLE 2.3-5
SUMMARY OF MASS AND COST DELTAS DUE TO PRIMARY EFFECTS OF A SMALL HLLV

PRIMARY EFFECT	MASS Investment MT	MASS Production MT	COST Invest $M	COST Production $M	COST Operations $M/YR	COMMENTS
o 7.5M SOLAR ARRAY BLANKETS						
o LEO Base	+12		+45		+1.03	Additional deployment equipment and crew
o GEO Base	+10		+87.6		+1.03	
o SPS		+232		+12.3		o Mostly Type A beam revisions (more battens)
o SMALLER ION THRUSTER PANELS						
o SPS						o Negligible effect
o EOTV						o Negligible effect
o LEO Base						o Negligible effect
o GEO Base						o Negligible effect

TABLE 2.3-5 (Continued)
SUMMARY OF MASS AND COST DELTAS DUE TO PRIMARY EFFECTS OF A SMALL HLLV

	MASS		COST			
	Investment	Production	Invest	Production	Operations	
PRIMARY EFFECT	MT	MT	$M	$M	$M/YR	COMMENTS
o MODULAR SLIP RING ASSEMBLY						
o SPS		+.9		+4		
o GEO Base	+15		+97.6		+2.06	o Added subassembly and test/checkout facilities, equipment, crew
o SMALLER AND MORE NUMEROUS CARGO PALLETS						
o LEO Base	+52		+171.8		+2.6	Primarily extra transporters and cargo tugs and associated crew
o GEO Base	+22		+78.8		+2.6	
o EOTV		+1		+1		
o SMALLER CREW MODULES						
o LEO Base	+596		+3026		+51.5	o 74 new crew members*
o GEO Base	+1394 to 2405		+6785 to 12575		+80.6	o 116 new crew members*

*Only half of these new crew members are in space—other half is on the ground.

TABLE 2.3-5 (Continued)

SUMMARY OF MASS AND COST DELTAS DUE TO PRIMARY EFFECTS OF A SMALL HLLV

PRIMARY EFFECT	MASS Investment MT	MASS Production MT	COST Invest $M	COST Production $M	COST Operations $M/YR	COMMENTS
o MORE HLLV'S						
o HLLV's						
o Launch/Recovery Site			-3049			o On-shore launch pads decrease cost dramatically (see Appendix 2-B)
o Ops Control			+189.7		+54.2	
o LEO Base			+34.6		+2.06	
o SMALLER OTV						o Deltas can be eliminated by redesign
o SMALLER ORBITAL PASSENGER MODULE						o Deltas can be eliminated by redesign

2.4 ESTIMATE OF DELTA ENVIRONMENTAL EFFECTS

2.4.1 Introduction

The objective of this task was to assess the environmental effects of the smaller and more numerous HLLV. These environmental effects include launch and reentry overpressure (sonic boom), launch facility noise, launch pad explosions, and effluent deposition in the upper atmosphere.

These environmental effects have been assessed for the baseline HLLV. The sonic boom, launch site noise, and launch pad explosion effects were reported in Reference 1. The effluent deposition effects were reported in Reference 2. The authors of these analyses (References 3, 4, 5 and 6) were asked to make judgments as to the delta environmental effects when comparing the smaller HLLV to the baseline HLLV. This report presents the results of these assessments.

2.4.2 Launch and Entry Overpressure

The sonic boom characteristics for the small HLLV during reentry are described below. The ascent sonic boom characteristics were not assessed as the ascent trajectory for the small HLLV is substantially different than that for the large HLLV. As the ascent sonic booms will occur over the ocean down-range from the launch site, it was judged that the ascent sonic overpressure characteristics do not need to be recomputed.

Sonic Overpressure Calculation—In Reference 1, the sonic overpressure of the SPS vehicles were computed using "the modified Witham equation" shown below:

$$P = \left(\frac{\sqrt{P_A P_G}}{h^{3/4}}\right)\left(K_R\right) \quad (M^2-1)^{1/8} \quad \left(\frac{d}{\ell^{1/4}}\right) K_v$$

where
ΔP = Bow shock overpressure in psf
P_A = Atmosphere pressure at vehicle altitude in psf
P_G = Atmosphere pressure at ground level in psf
h = Perpendicular distance from flight path in feet
K_R = Reflection factor (usually about 2.0)
M = Vehicle Mach number
d = Vehicle diameter
ℓ = Vehicle length
K_v = Vehicle volume shape factor ($.54 \leq Kv \leq .87$) ; assumed to be 0.8

For our purposes in the analysis of the small HLLV, the only factor that will be different from those used for the large winged HLLV analysis is the $d/l^{1/4}$ factor. It was judged (Reference 3) that the under flight track overpressures for the small HLLV could be scaled from the large HLLV data.

$$\text{Scale factor} = \frac{(d/\ell^{1/4}) \text{ Small HLLV}}{(d/\ell^{1/4}) \text{ Large HLLV}}$$

$$= \frac{(12.5/112.7^{1/4})}{(15.25/140.73^{1/4})} = \frac{3.836}{4.427}$$

$$= .8665 \text{ (use .87)}$$

Sonic Overpressure Patterns—The overpressure along the vehicle flight track predicted by the modified Witham equation is shown in Figure 2.4-1. These overpressures were used together with the data from program TEA-251 to determine sonic boom overpressure patterns lateral to the ground track, see Figures 2.4-2 and 2.4-3.

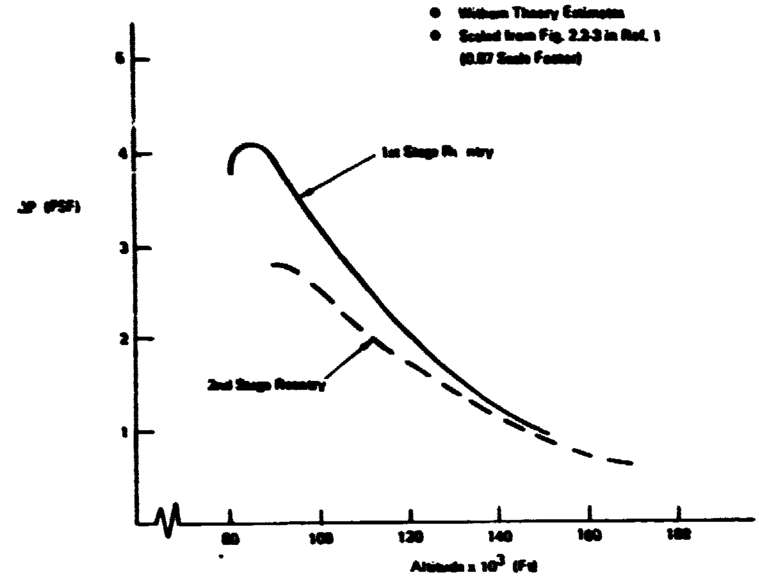

Figure 2.4-1. Ground Sonic Boom Overpressures Under Flight Track—Small HLLV Reentry

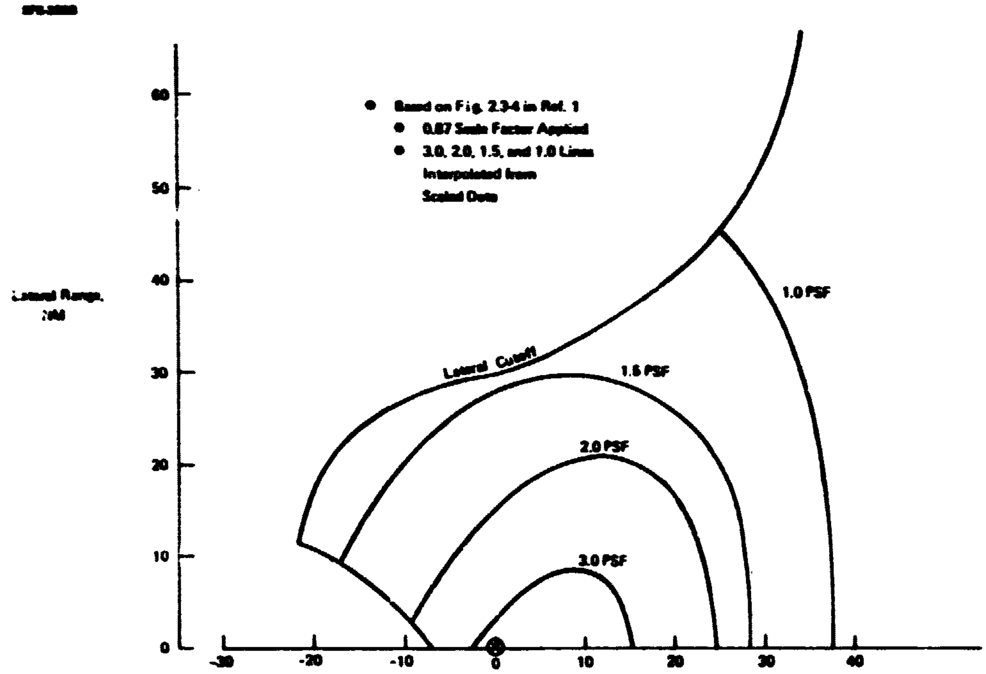

Figure 2.4-2. Small HLLV Booster Reentry Sonic Boom Overpressure

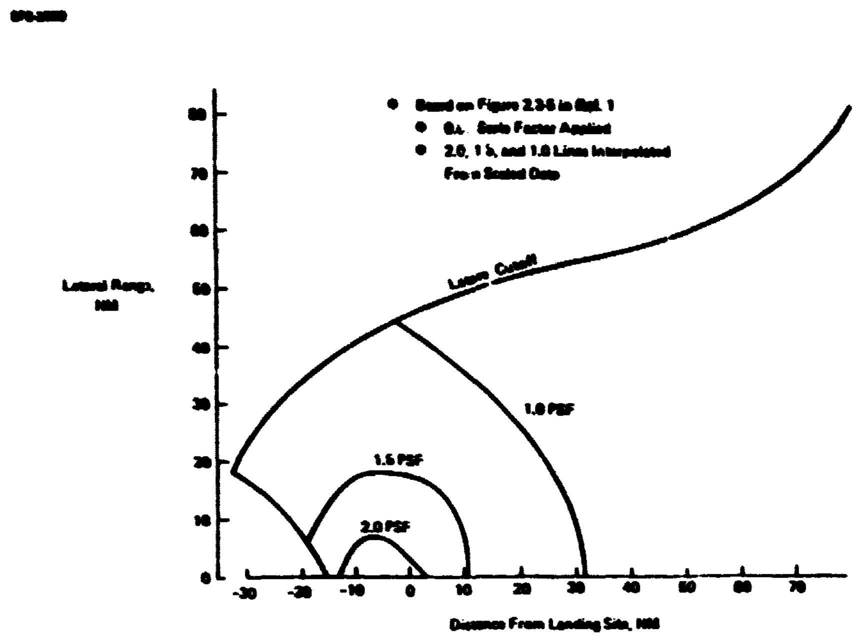

Figure 2.4-3. Small HLLV Second Stage Reentry Sonic Boom Overpressure

The reentry sonic boom pressure signatures (ΔP vs time) at selected locations are not scalable (Reference 3).

Effects of the Sonic Overpressures—In Reference 1, the physical and behavioral effects on humans of sonic overpressures and the structural damage effects of sonic booms were enumerated. From this data, it was recommended that the maximum allowable overpressure of 2.0 psf outside of the government reservation perimeter shall not be exceeded.

What this translates to for the small HLLV is that the perimeter of the government reservation must be at least 25nm from the landing site on the line of approach (based on Figure 2.4-2) and at least 13nm downrange (based on Figure 2.4-3). The corresponding exclusion ranges for the large HLLV were 27nm and 17nm respectively.

2.4.3 Launch Noise

Launch Noise Calculation—In Reference 1, the launch noise was predicted by a procedure that utilizes the basic jet noise generation influencing parameters (jet velocity, density, mass flow, temperature and nozzle area). The small HLLV uses 6/16 of the number of the same engines that were used in the original analysis. The scaling factor is 10 log 6/16 = -4.26 db. For convenience, it was recommended (Reference 4) that -5 db be used (the predictions are only accurate to 0.5 db) to adjust the data plots found in Reference 1.

Launch Noise Data—The predicted launch Overall Sound Pressure Level (OASPL) contour map for the small HLLV is shown in Figure 2.4-4. The predicted Perceived Noise Level (PNL) contour is shown in Figure 2.4-5. These contour maps represent the maximum noise emitted by the launch vehicle at the site. These noise predictions are limited to the static case where the vehicle is considered to have no forward motion.

As a measure of relative comparison, the building damage noise limit (as suggested on the basis of a literature survey) of 147 db OASPL is prescribed. For habitation, the PNL levels should not exceed 108 db.

Figure 2.4-6 shows the OASPL and PNL levels for the small HLLV as a function of radial distance along the ground surface ($= 90^{o}$). From extrapolation of this curve, it can be seen that the maximum OASPL level for building damage occurs at about 400 ft (for the large HLLV, the corresponding location was 1000 ft). The PNL limit of 108 db takes place at 21,000 ft (for the large HLLV, the corresponding location was at 32,000 ft).

Figures 2.4-7 through 2.4-9 present the polar plot of the predicted OASPL for 1000, 10,000, and 100,000 ft distances. The PNL predictions for the same distances are shown in Figures 2.4-10 through 2.4-12. Figures 2.4-13 to 2.4-15 show the sound spectrum along the ground plane for the above distances.

2.4.4 Explosive Hazard Due To The Propellant Combinations

The explosive hazard of the propellant combinations used in the small HLLV was estimated using the procedures used for the large HLLV, see Reference 1. To adjust the data from this reference it was necessary to define the scaling factor shown below (Reference 6):

<u>Small HLLV</u>

Booster: $LO_2 + LCH_4 = 4.823 \times 10^6$ lb x .2 = 964,600 lb of TNT equivalent

Second Stage: $LO_2 + LH_2 = 2.491 \times 10^6$ lb x .6 = <u>1,494,600</u> lb. of TNT equivalent

$2{,}459 \times 10^6$ lb. of TNT equivalent

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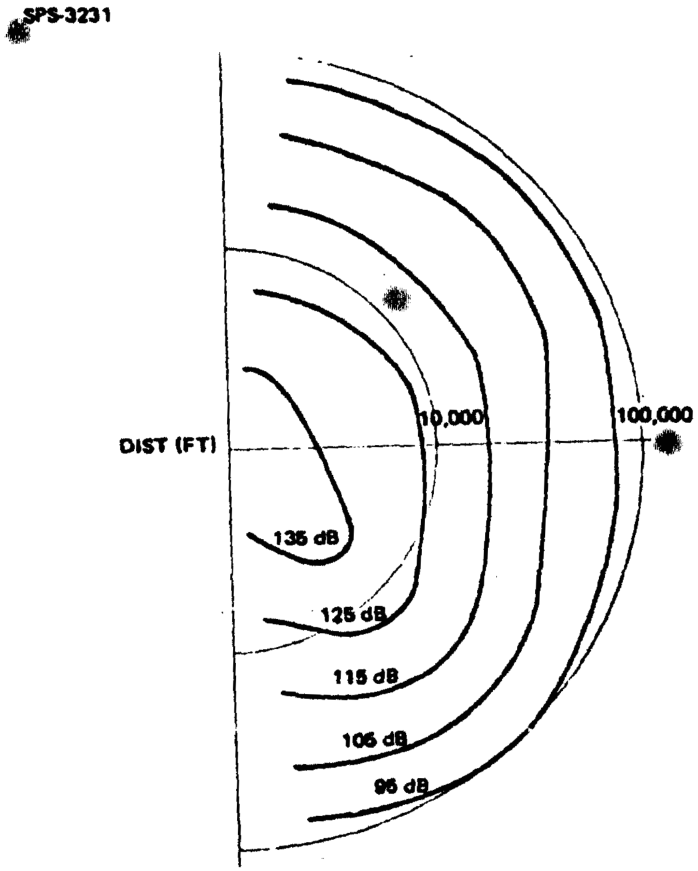

Figure 2.4-4. SPS Predicted Overall Sound Pressure Levels—OASPL-dB

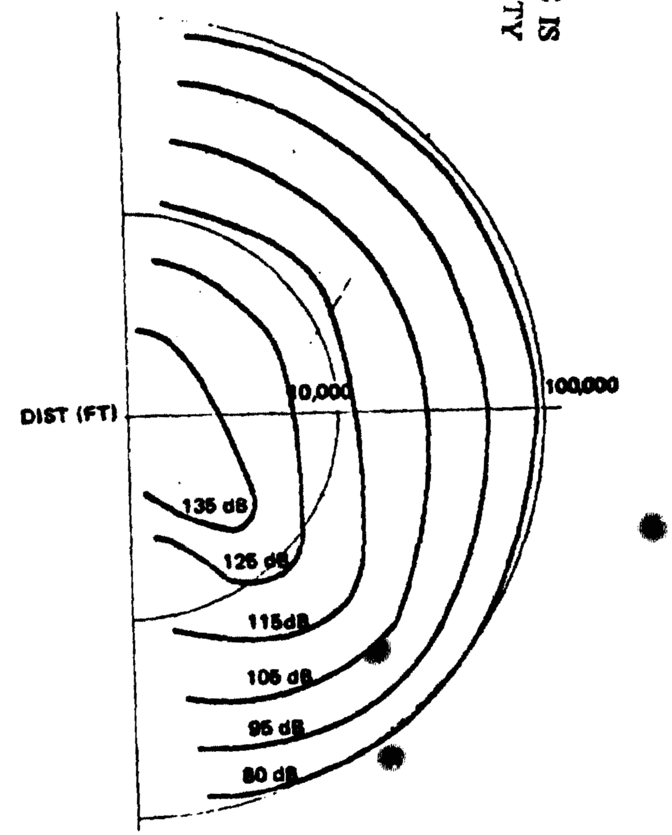

Figure 2.4-5. SPS Predicted Perceived Noise Levels—PNL-dB

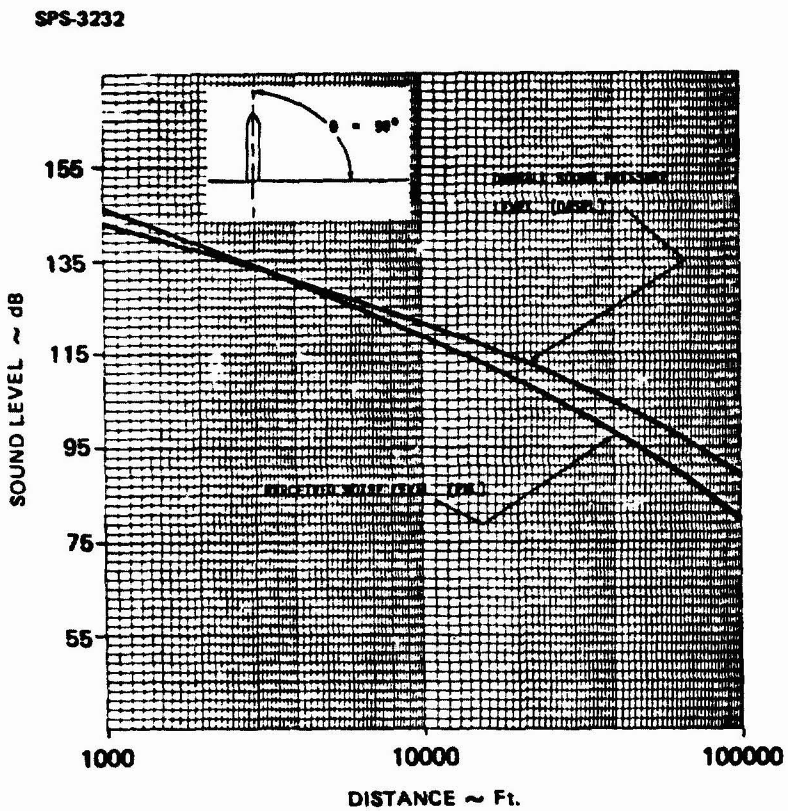

Figure 2.4-6. Launch Site Ground Surface Noise Levels (θ = 90°)

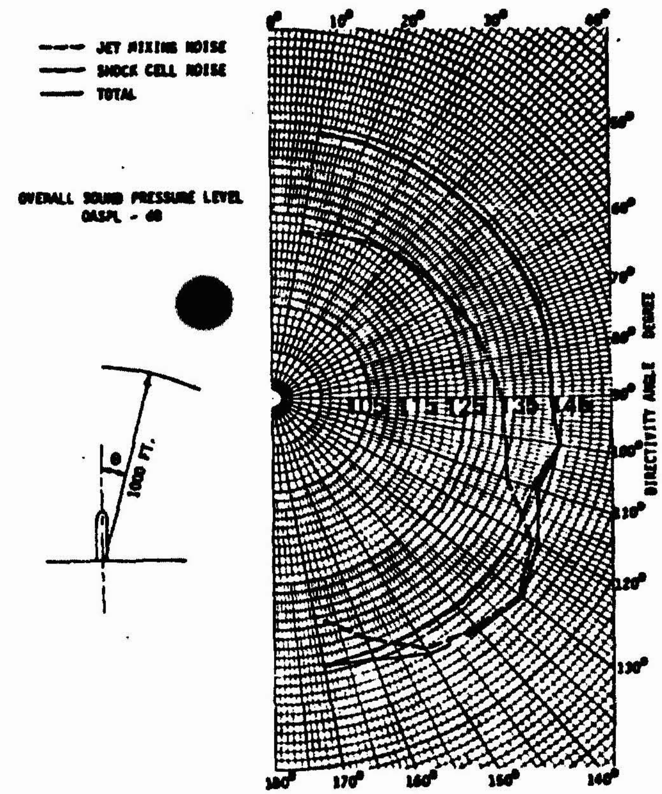

Figure 2.4-7. SPS Launch Vehicle Overall Sound Pressure Level-dB (1,000 ft. Sideline Distance)

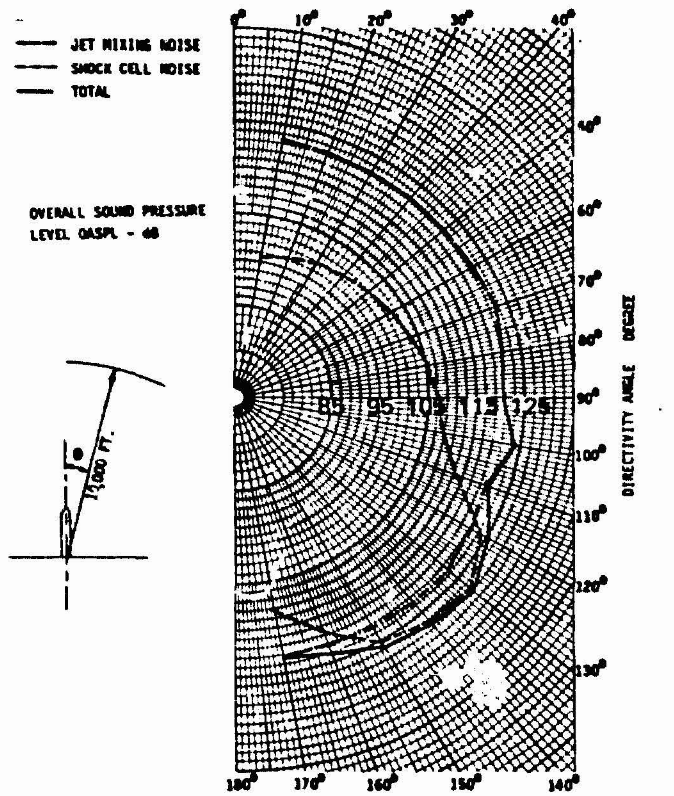

Figure 2.4-8. SPS Launch Vehicle Overall Sound Pressure Level-dB (10,000 ft. Sideline Distance)

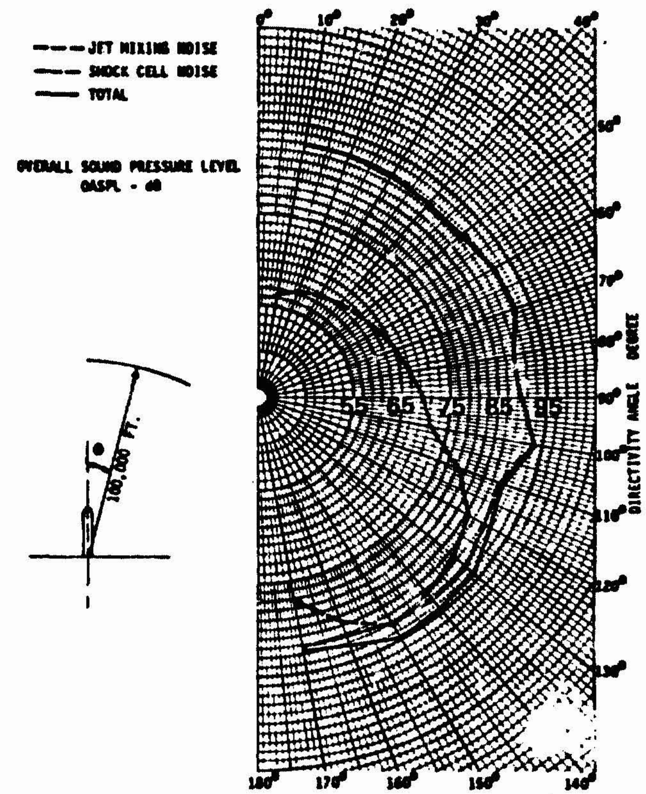

Figure 2.4-9. SPS Launch Vehicle Overall Sound Pressure Level-dB (100,000 ft. Sideline Distance)

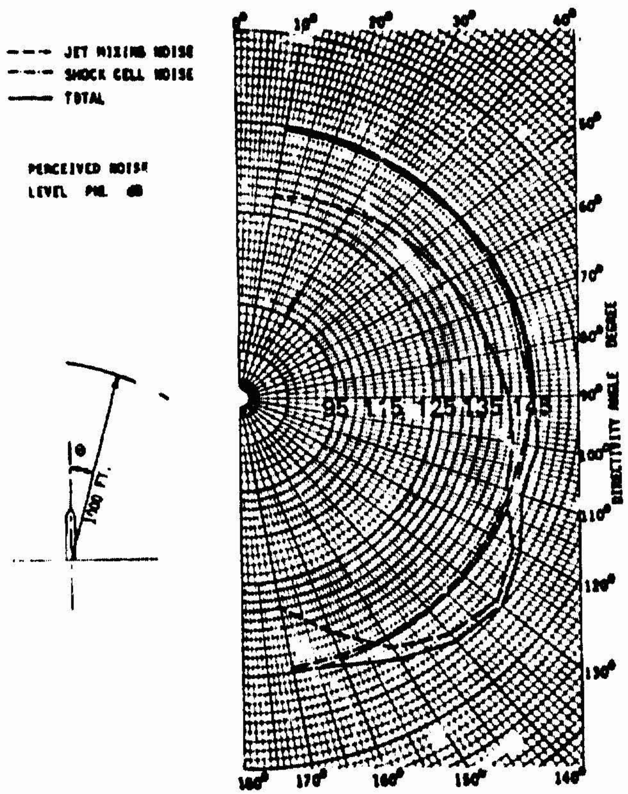

Figure 2.4-10. SPS Launch Perceived Noise Level-dB (1,000 ft. Polar Distance)

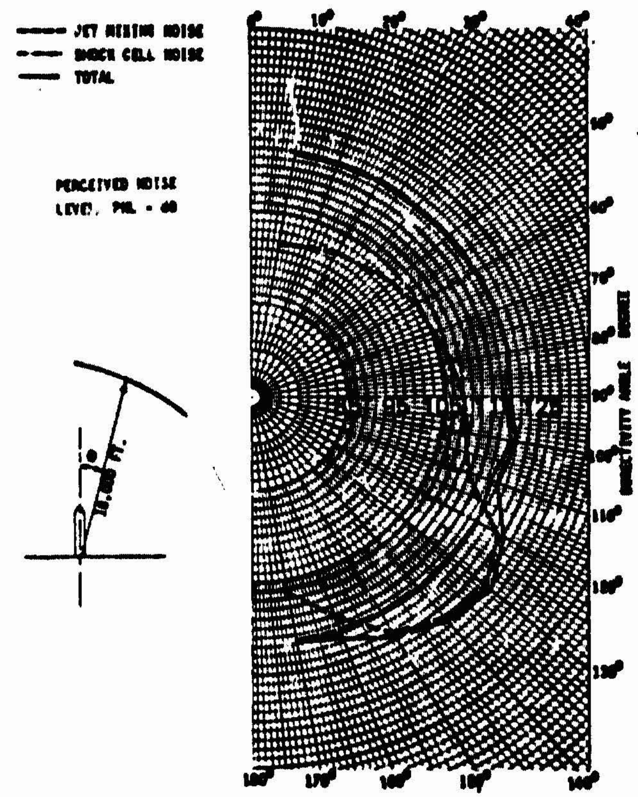

Figure 2.4-11. SPS Launch Perceived Noise Level-dB (10,000 ft. Polar Distance)

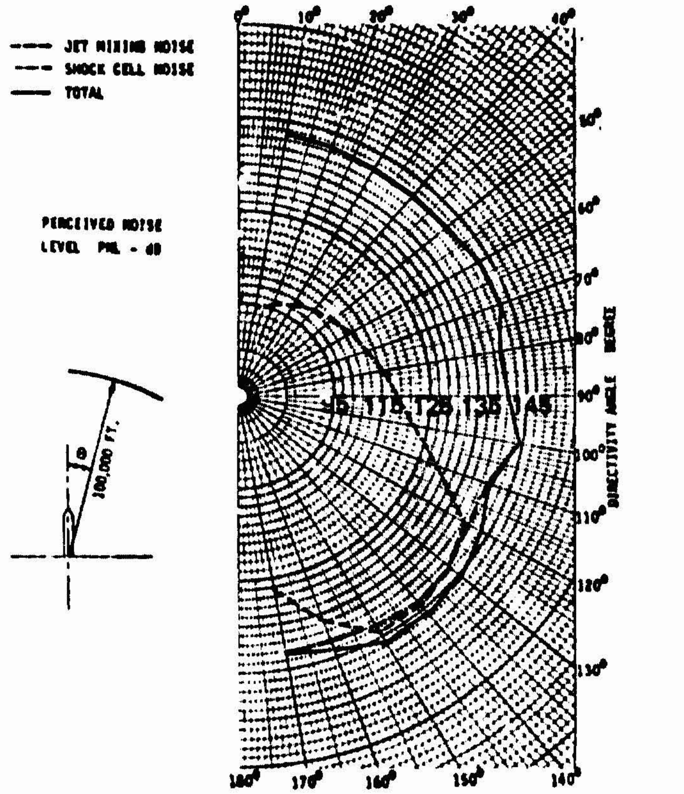

Figure 2.4-12. SPS Launch Perceived Noise Level-dB (100,000 ft. Polar Distance)

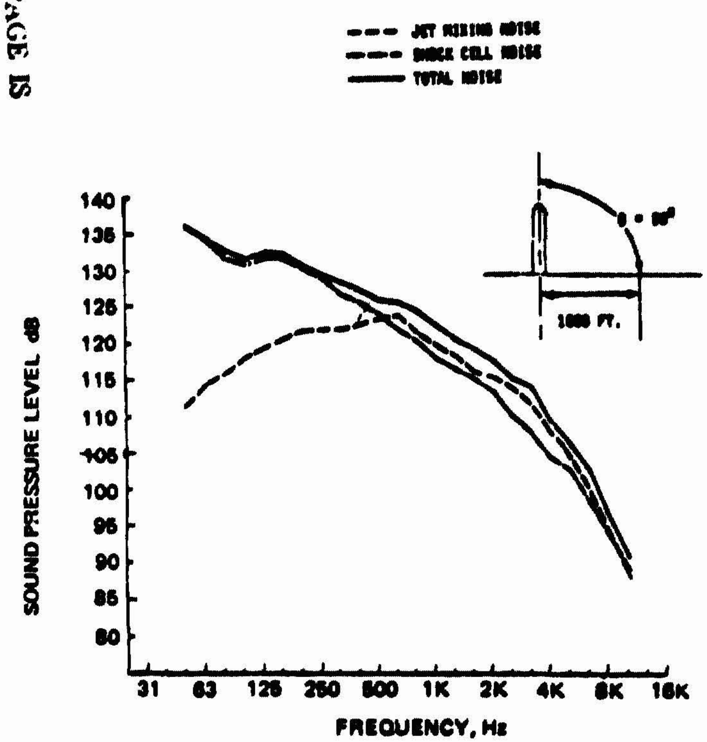

Figure 2.4-13. SPS Sideline Noise Spectrum—1,000 ft. Sideline Distance

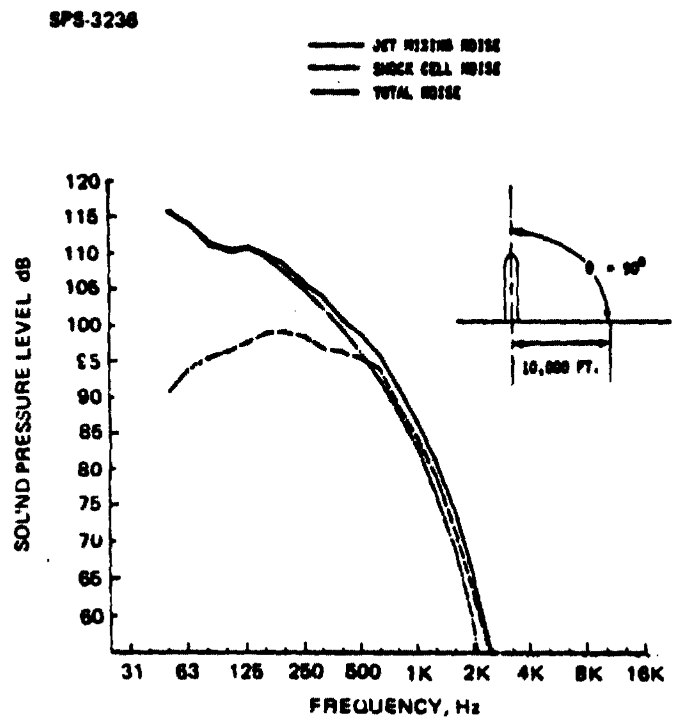

Figure 2.4-14. SPS Sideline Noise Spectrum— 10,000 ft. Sideline Distance

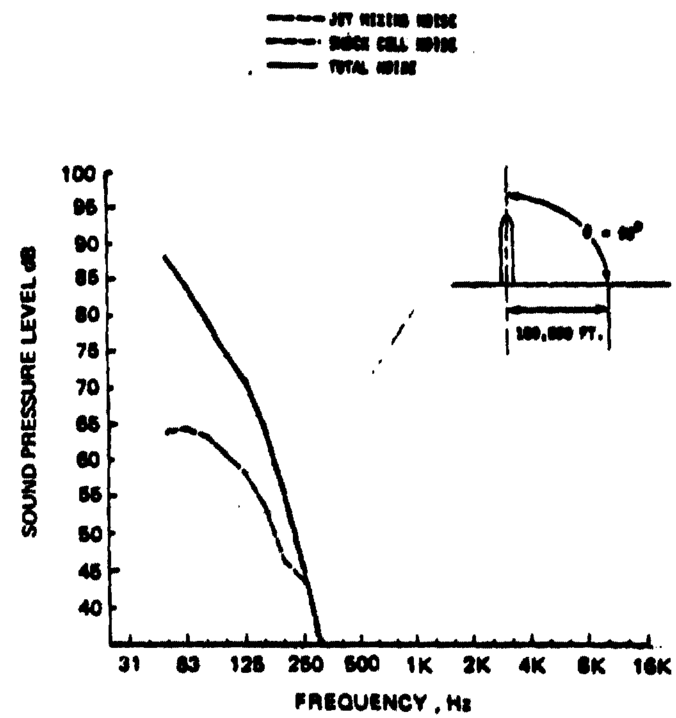

Figure 2.4-15. SPS Sideline Noise Spectrum— 100,000 ft. Sideline Distance

$$\text{Scaling Factor} = \frac{\sqrt[3]{2.459\times10^6}}{\sqrt[3]{6.200\times10^6}} = 0.737$$

Large HLLV
TNT equivalent

The predicted overpressures from an on-pad explosion of the small HLLV are shown in Figure 2.4-16. Using the same 0.75 psi overpressure limitation as was used for the large HLLV, the minimum pad separation distance for the small HLLV becomes 1.32nm (8000 ft). The corresponding pad separation distance for the large HLLV was 2nm (12,156 ft).

2.4.5 Effluent Deposition in the Upper Atmosphere

In Reference 2, the deposition of H_2 and H_2O into the upper atmosphere by the large HLLV's was assessed. The corresponding effects for the smaller HLLV was estimated from this data (Reference 5).

For the large HLLV, there were approximately 8 flights per week. For the smaller HLLV, there will be 35 flights per week (for the corresponding year of SPS construction). There will, therefore, be 35/8 = 4.38 times as many flights per week.

The second stage propellant mass for the large HLLV was 5.1×10^6 kg. For the small HLLV, the corresponding mass is 1.13×10^6 kg. Therefore, each of the small HLLV's will inject 51% as much of the effluent as the large HLLV.

The net effect will be 1.73 times as much effluent injected into the upper atmosphere each week by the small HLLV when compared to the large HLLV. However, this may not be as bad as it may seem.

The density of effluents for each of the smaller HLLV's will be approximately half of that for the larger HLLV's. Furthermore, these effluents will be spread along a smaller diameter line source for each vehicle flight. The speed of diffusion will, therefore, be decreased due to the more rapidly decreasing concentration gradients. This will allow more time for favorable chemical reactions to occur before the effluents diffuse to the ionosphere.

As with the previous analysis (in Reference 2), the provision must be made that these predictions are very preliminary in nature in that some very important simplifying assumptions have been made to allow the analysis to be done. More detailed analyses should be done as there may be some subtle effects that may either harm or help the effluent problem.

2.4.6 Summary

In this report, we have presented the results of a comparative assessment of the environmental effects of the small HLLV versus those of the baseline large HLLV.

The series-burn stack height is commensurate with that of Saturn V, indicating that present facilities can be used in the developmental phase. The operational, high-launch-rate, ground handing system will probably move the empty vehicles on their own landing gear, mate in the horizontal position at the launch pad, and use a strong-back tilt-up launcher.

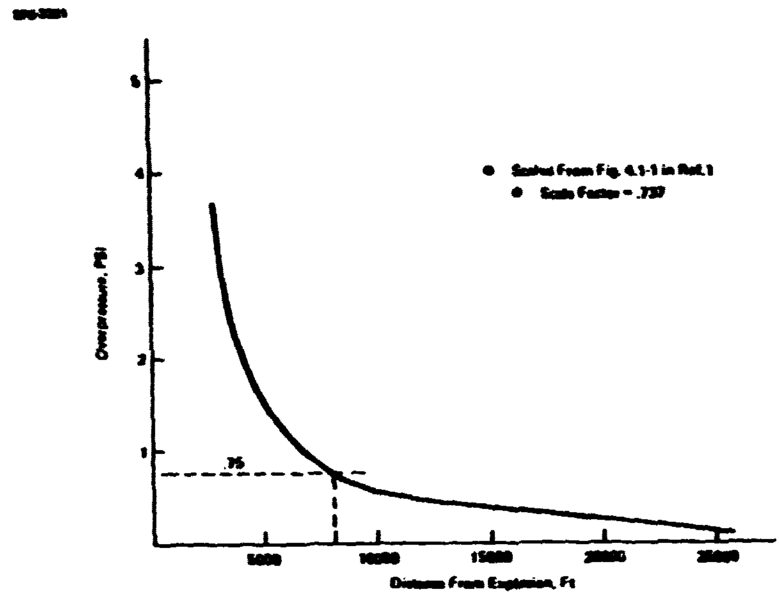

Figure 2.4-16. Predicted Overpressures from On-Pad Explosion (Small HLLV)

Sonic Boom—The second stage vehicle reentry will be the source of the most severe sonic booms at the launch and recovery site. The recommended sonic boom overpressure at the boundary of the government reservation is 2.0 psf. Figure 2.4-17 shows that this 2.0 psf boundary for the small HLLV is somewhat less than that required for the large HLLV.

Launch Noise and Blast—The launch noise levels for the small HLLV will be substantially less than that for the large HLLV. Figure 2.4-18 shows that adjacent structures can be 60% closer to the small HLLV launch pads when noise level structural damage is considered. Figure 2.4-19 shows the minimum pad separation required based on an on-pad explosion. The pads can be over 4000 ft closer together than was required for the large HLLV. This figure also shows that the minimum distance to habitable areas can be 12000 ft closer, based on human noise exposure limitations.

Upper Atmosphere Effluents—The small HLLV will deposit 1.71 times as much effluent into the atmosphere per week as the large HLLV. However, this increase may be substantially offset by a slower rate of diffusion that will allow the effluents to be chemically decomposed into non-harmful constituents.

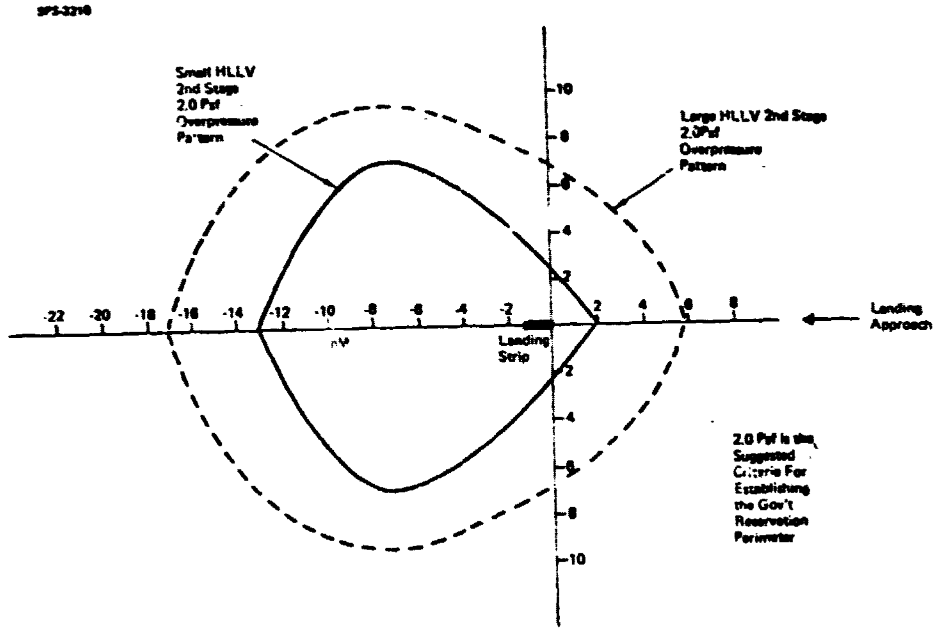

Figure 2.4-17. Minimum Distance from a Launch Pad to Adjacent-Structures Based on Noise Level Criteria

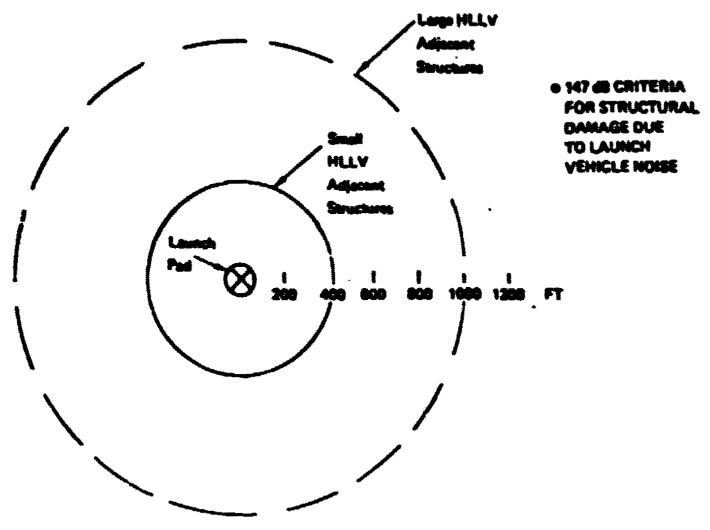

Figure 2.4-18. Minimum Distance from Launch Pad to Adjacent Structures Based on Noise Level Criteria

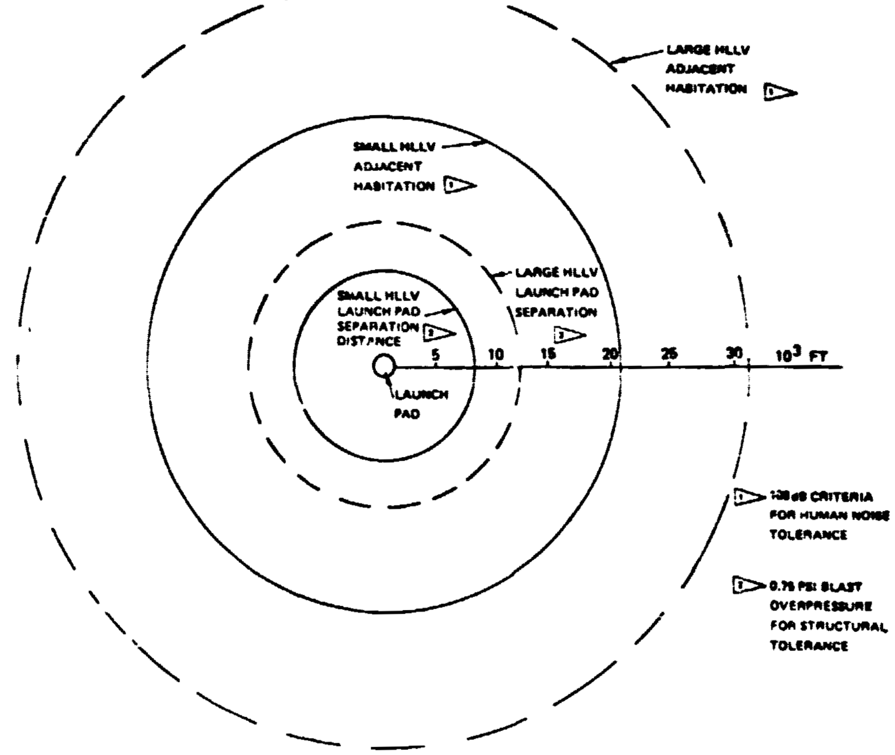

Figure 2.4-19. Minimum Distance from Launch Pad to Adjacent Habitable Areas and to Adjacent Launch Pads

2.5 COST ANALYSIS

It was estimated that the small HLLV would inherit several subsystems and technologies that could be used with suitable modifications. The principal ones are the following:

FROM SHUTTLE

- o ORBITER MAIN ENGINES
- o THERMAL PROTECTION SYSTEM
- o AVIONICS & POWER
- o CREW SYSTEMS
- o REACTION CONTROL SYSTEM

FROM OTV

- o ORBIT MANEUVER ENGINES

FROM MILITARY OR COMMERCIAL AIRCRAFT

- o BOOSTER FLYBACK ENGINES

Cost estimating factors are summarized in Table 2.5-1. The top part of the table indicates the DDT&E costs. The center part shows the commonality credits from the shuttle and OTV, and the bottom summarizes the theoretical first unit costs and learning slopes for vehicle production.

The development costs figures from the Table 2.5-1 are shown in pie chart fashion in Figure 2.5-1. Note that totals are also indicated. The relatively small main engine contribution for the orbiter results from the assumption that the space shuttle main engine is to be used essentially as is.

The principal contributors to cost per flight are enumerated in Table 2.5-2.

The scenario indicated a nominal launch rate of 1500 flights per year. The program average cost per flight is shown in pie chart fashion in Figure 2.5-2. As was true for the reference HLLV, flight hardware for amortization of vehicles and spares and maintenance dominates the total. Ground system and operations include those people directly involved in vehicle turnaround operations. Site manpower and program support are indirect people chargeable to launch operations. Tooling sustaining reflects a 10% a year figure based on initial tooling costs. Finally, propellants were costed as they were costed for the reference HLLV.

The delta costs between the small HLLV and the large reference system are summarized in Table 2.5-3 page. Satellite design changes resulted in increased costs for the space construction systems that were reflected as nonrecurring invesment costs in hardware. The necessity to use smaller crew modules results in a DDT&E savings, but an investment increase from the need to buy more of the smaller modules. Transportation includes direct DDT&E savings on the smaller launch vehicle, savings resulting from less complex facilities and increase in the fleet investment and in the HLLV factory and savings resulting from less development activity on shuttle derivatives as a result of having the small heavy lift launch vehicle. It may be noted that the large increase in HLLV factory and tooling costs probably, in part, reflects an underestimate in tooling for the large HLLV. The cost model has been updated since the original figures were developed and now reflect higher tooling costs. In the recurring column, results include the cost of SPS hardware under SPS, the cost of transporting the additional SPS mass under Transportation, and the cost of construction operation in the third column. Recurring cost for the

Table 2.4-1. Small HLLV Cost Summary

	DDT&E	
	BOOSTER	ORBITER
AIRFRAME	1977	3120
MAIN ENGINE	1619	215
AUXILIARY PROPULSION	151	26
SUBSYSTEMS	316	381
GROUND & FLIGHT TEST VEHICLES	704	525

	ORBITER COMMONALITY CREDITS (DDT&E)	
MAIN ENGINE	0.95	(SSME)
OMS	0.8	(OTV)
RCS	0.5	(SHUTTLE)
ELECTRIC POWER, AVIONICS, EC/LSS	0.7	(SHUTTLE)

	PRODUCTION			
	BOOSTER		ORBITER	
	TFU	SLOPE	TFU	SLOPE
AIRFRAME & SUBSYSTEMS	178	.85	187	.85
MAIN ENGINE (6 PER STAGE)	32	.90	18	.90
AUXILIARY PROPULSION	4.5 (4 REQ'D)	.88	5.1	.88

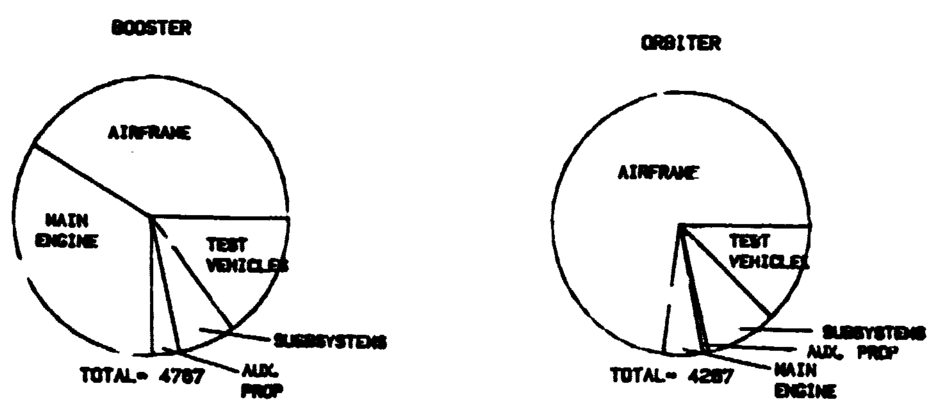

Figure 2.5-1. Small HLLV Development Cost

Table 2.5-2. Cost per Flight (1500/yr)

ITEM	COST IN MILLIONS (79$)
PROGRAM SUPPORT	.113
FLIGHT HARDWARE	2.359
GROUND SYSTEM & OPS	0.35
TOOLING SYSTEMS	0.18
PROPELLANT	0.617
SITE MANPOWER	0.612
	4.231

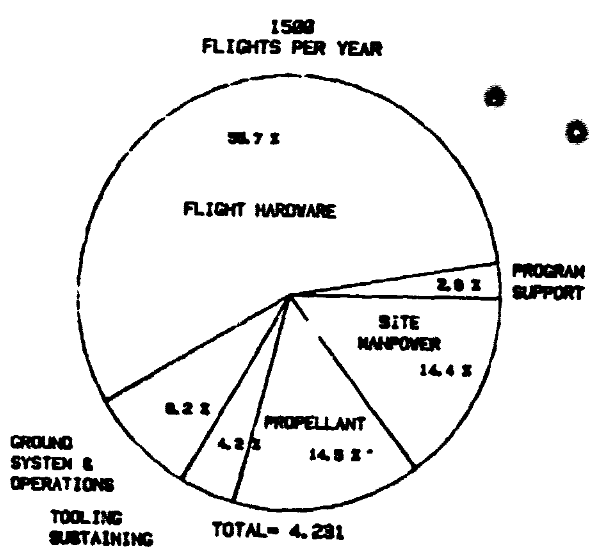

Figure 2.5-2. Small HLLV Cost per Flight

Table 2.5-3. Delta Cost Summary—Small HLLV

[illegible]

	NONRECURRING	RECURRING SPS	RECURRING TRANSPORTATION	RECURRING CONSTRUCTION
SATELLITE DESIGN CHANGES	230 (BASE CHANGES)	16.3	90.4	4.12
CARGO LOGISTICS	250.1	—	—	5.2
SMALLER CREW MODULES		—	—	132.1
DDT&E	-2521 [1]			
INVESTMENT	3925 + 34.4			
TRANSPORTATION		—	1040 (HLLV) -400 (PLV)	—
DDT&E	-3075			
FACILITIES INVESTMENT	-3049			
FLEET INVESTMENT	790 [2]			
HLLV FACTORY	1619			
LESS SHUTTLE MODS	-3204			
TOTAL	-5000.6	16.3	730.4	141.42
		TOTAL = 887		

[1] INCLUDES CREDIT FROM DEMONSTRATION PHASE

[2] TOOLING UNDERESTIMATED FOR LARGE HLLV?

small HLLV is higher than for the large one, but the small HLLV also accomplishes crew rotation from Earth to low Earth orbit, resulting in a savings. The net recurring result is 887 millions per year, about 440 millions per SPS, or roughly 3% increase per SPS.

2.6 CONCLUSIONS/RECOMMENDATIONS

In summary, the small HLLV has positive features and some negative features. Table 2.6-1 summarizes these positive and negative features. In general, the positive features outweigh the negative features and it is recommended that the small HLLV be adopted as an SPS reference system.

<u>POSITIVE</u>

- o LESS NONRECURRING COST: MORE COMMONALITY WITH SHUTTLE
- o REDUCED NOISE & SONIC OVERPRESSURE
- o LESS FACILITIES COST: OFFSHORE PADS NOT NEEDED
- o SIZE APPROPRIATE FOR ALTERNATIVE MISSIONS
- o CREW AS WELL AS CARGO DELIVERY

<u>NEGATIVE</u>

- o SLIGHTLY HIGHER RECURRING COST
  - . GREATER NUMBER OF CONSTRUCTION CREW
  - . MORE PROPELLANT CONSUMED
- o MORE FREQUENT FLIGHTS
- o MORE EFFLUENT DEPOSITED IN UPPER ATMOSPHERE

Figure 2.6-1. Small HLLV Net Effects

3.0 SHUTTLE-DERIVED SPS TRANSPORTATION

The goal of the shuttle-derived SPS transportation system concept was to minimize transportation development cost. The question related to this goal was determination of the recurring cost for SPS production if this transportation system were adopted.

3.1 Initial Concept

The concept involves use of shuttle orbiters and external tanks both for Earth-to-orbit and for orbit-to-orbit transportation. In order to reduce costs and increase performance, a new booster is to be designed and developed. This concept was developed by Jim Akkerman of the Johnson Space Center. An initial configuration was provided as a part of the Phase III task statements. The configuration had certain known problems. First of all, very little volume was available for SPS hardware payloads. These hardware payloads are relatively low in density and require a large-volume payload bay to achieve efficient transportation operations. Further, the original concept included a redesign of the satellite, fairly complex construction operations, and raised certain questions as to whether the large sections of satellite built at low Earth orbit could be transported to GEO. Thirdly, accommodations for crew delivery for LEO to GEO were not provided. Finally, the system included a ballistic booster. Earlier studies of ballistic versus winged boosters had indicated that winged systems would provide lower transportation costs due to more rapid turnaround.

A revised configuration was developed that included a redesign of the external tank and the use of a flyback booster. It had also been suggested that the orbiter be redesigned to provide increased payload accommodations. This, however, appeared to be in conflict with the desired objective of minimizing development costs. If one were to redesign the orbiter and provide a new booster, one would, in effect, have a small heavylift launch vehicle. (That option was reported in the previous section of this report).

Figure 3.1-1 shows the principal features of the modified system. Cargo space is provided in the external tank. The shuttle cargo bay provides sufficient volume for personnel accommodation. The flyback booster and interstage structure provide for launch of the shuttle and external tank to the proper staging conditions.

Cargo is launched to low Earth orbit with the configuration illustrated. Some of the external tanks with cargo space are to be used for orbit-to-orbit transportation. These are provided with better thermal insulation for roughly a week's stay time in low Earth orbit. Additional launches with relatively conventional external tanks bring propellant to low Earth orbit to fill the orbit transfer ET systems. The relatively high performance of the large flyback booster allows the system to arrive in orbit with substantial propellant remaining in the external tank. This is then transferred to the orbit transfer ET's until they are fully loaded with propellant.

In order to provide an adequate mass fraction for orbit transfer and allow the shuttle orbiter to go along as a propulsion system and crew transfer system, several external tanks are docked together end-to-end to provide a very large orbit transfer system with great propellant mass.

The principal featurs of the revised system are tabulated in Table 3.1-1. Note that three types of external tanks are required. The cargo for launch from Earth to orbit is housed internally to the external tank payload bay. For orbit transfer, this is not necessary and

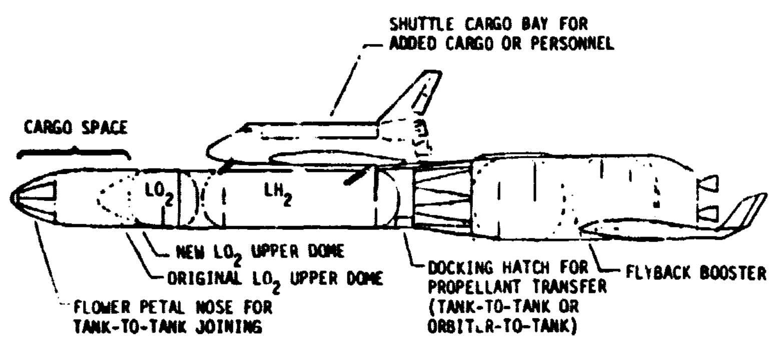

Figure 3.1-1. Modified Shuttle SPS Transportation System Cargo Launch Configuration

Table 3.1-1. Features of Revised System

- o CARGO SPACE IN ET ALLOWS DELIVERY OF CARGO TO GEO & ALL CONSTRUCTION AT GEO.
- o ADEQUATE VOLUME CAN BE PROVIDED.
- o ORBITER BAY AVAILABLE FOR PERSONNEL
- o THREE ET VERSIONS
  - (1) "REGULAR" - PROPELLANT DELIVERY TO LEO - MODIFIED ONLY FOR PROPELLANT ACQUISITION AND TRANSFER
  - (2) CARGO TO LEO - CARGO BAY ADDED
  - (3) LEO-GEO
    - o CARGO BAY
    - o FLOWER PETAL NOSE
    - o BETTER INSULATION

cargo brought to Earth orbit by those external tanks not configured for orbit transfer will be stored externally to the orbit transfer ET's for the orbit transfer.

3.2 Analysis

A number of questions were raised as to how to configure this system for minimum cost. The three principal variables are the booster size and attendant staging velocity, booster flyback optimization, and the number of external tanks to be provided for each transfer flight. Crew accommodations in the orbiter were a secondary question.

In order to conduct the optimization analysis, the ISAIAH Systems Modeling Software System was employed. The ISAIAH software, in effect, allows one to very quickly develop a computer program to analyze a complex systems model by standardizing those things that normally cause most of the difficulty in developing computer models. Table 3.2-1 summarizes the features of this system.

The ISAIAH System operates with the computer network at the Boeing Kent Space Center. The system is accessible through remote terminals and all card image files are maintained on disk files to avoid card deck handling. The software runs on a large IBM mainframe and plot files are transmitted to the interactive computer graphics facility for rapid plotting of results. Figure 3.2-1 illustrates the computer network.

The systems model is summarized in Figures 3.2-2 and 3.2-3. The segment of the model shown in Figure 3.2-2 includes the booster flyback optimization with principal variables being the booster wing area, dry inerts, and the booster propellant load and staging velocity. The iterations implied in the network are handled automatically with the Isaiah software.

The analysis of the upper stages is diagramed in Figure 3.2-3. As the ideal staging velocity increases, the upper stage injected mass increases thus increasing the cargo mass and the propellant deliverable. However, as the ideal staging velocity increases, larger and larger boosters are required so one would expect a minimum cost point.

The next several figures shown modeling inputs that were incorporated into the model as lookup tables. The estimated relationship of booster wing mass to the booster mass and booster-wing area is shown in Figure 3.2-4. This is a key relationship for establishing the flyback optimization.

The staging relative path angle decreases with increasing staging velocity as shown in Figure 3.2-5; the path angle is important in establishing flyback range.

Shown in Figure 3.2-6 is the relationship of relative staging velocity to ideal staging velocity.

The flyback range is composed of two principal components: the range at staging and the coast range after staging. Shown in Figure 3.2-7 is the range at staging as a function of ideal staging velocity. On the next Figure (3.2-8), the coast flyback range as a function of path angle and inertial staging velocity are shown.

The booster theoretical first unit cost is modeled as dependent upon the booster wet inert weights (booster inerts including residual ascent propellants but not including flyback propellant). The model included learning curve relationships to allow the booster average unit cost to be computed from the theoretical first unit cost. The TFU is shown in Figure 3.2-9.

Table 3.2-1. ISAIAH Description

- STANDARDIZED, STRUCTURED PROCEDURE AND SOFTWARE SYSTEM FOR INTERRELATIONSHIPS AND SENSITIVITY ANALYSIS
  - MODELING METHODOLOGY
  - INPUT LANGUAGE
  - INTERNAL LOGIC
  - DIAGNOSTICS
  - OUTPUT FORMATTING
  - PLOT ROUTINES

- NINETY PERCENT OF THE CODE AND 95% OF THE TROUBLE IN A LARGE COMPUTER PROGRAM IS INPUT, OUTPUT, LOGIC STRUCTURE, AND FILE HANDLING. THE RATIO IS SOMEWHAT WORSE IF COMPUTER GRAPHICS IS USED. WITH THE ISAIAH METHODOLOGY ALL OF THIS STUFF IS ALREADY THERE AND DOESN'T NEED CHANGING.

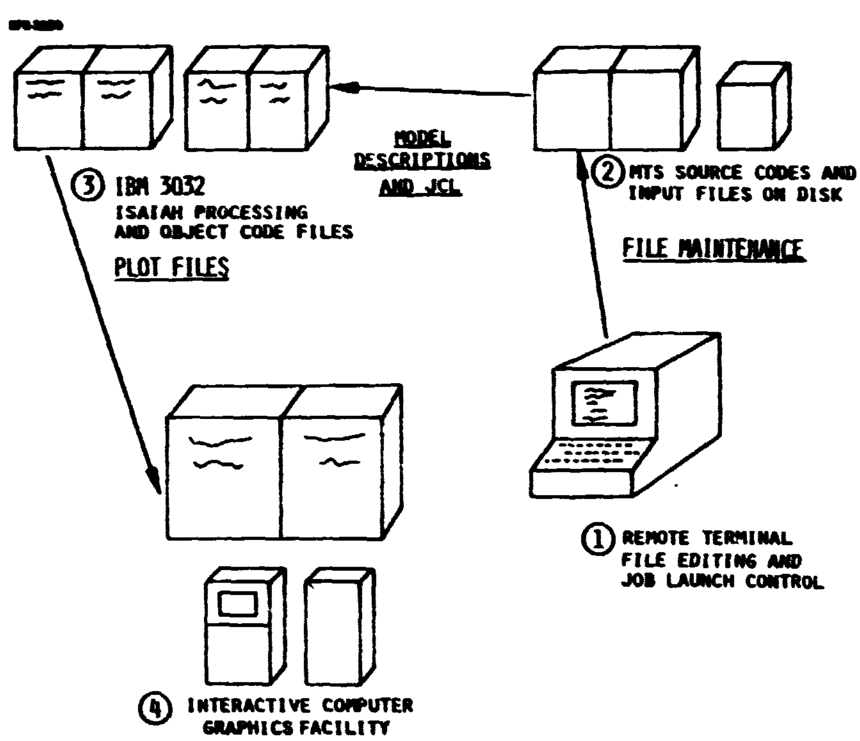

Figure 3.2-1. ISAIAH Computer Hookup

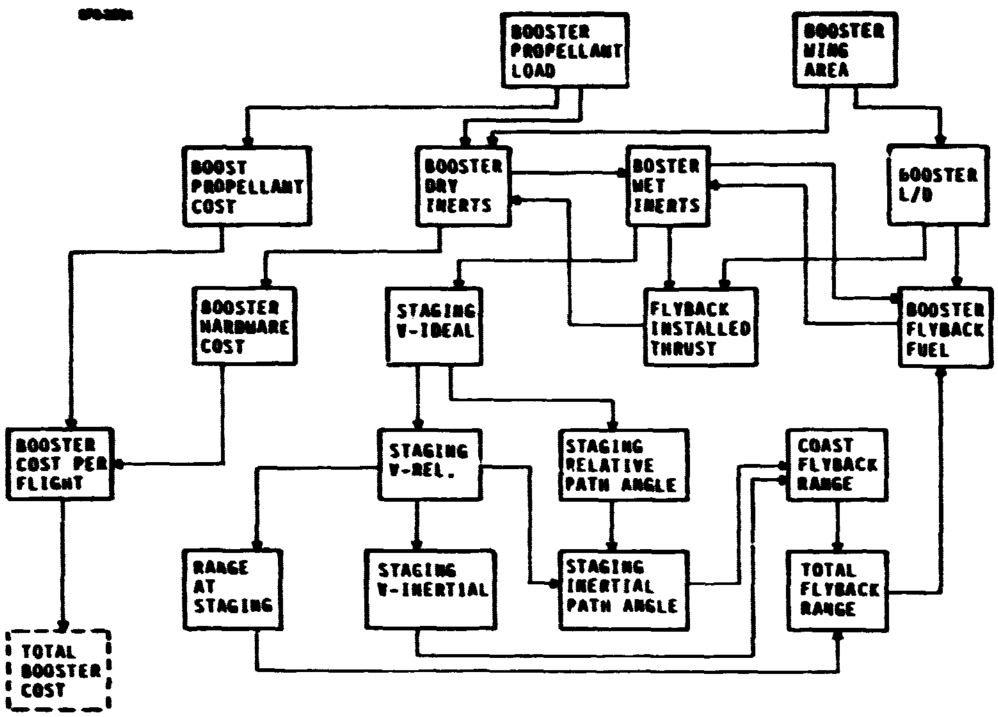

Figure 3.2-2. Shuttle-Derived System Optimization (Booster)

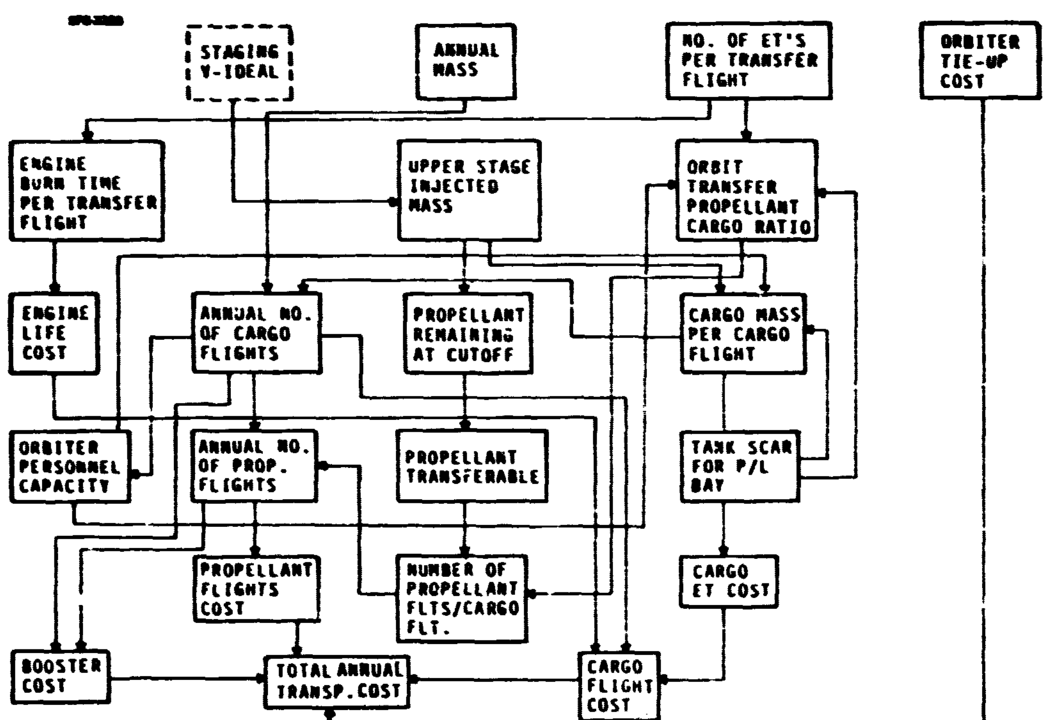

Figure 3.2-3. Shuttle-Derived System Optimization (Upper Stages and Total)

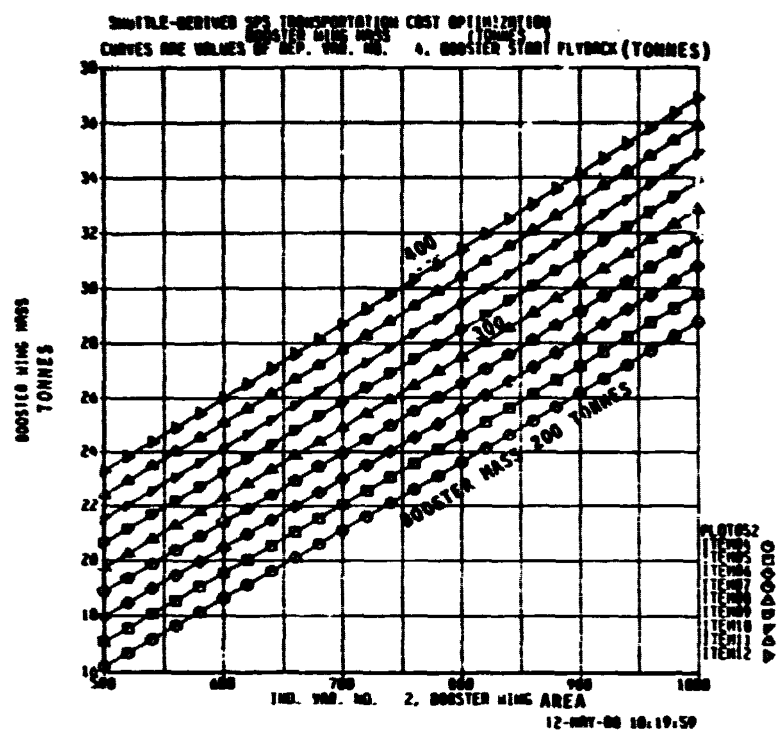

Figure 3.2-4. Model Inputs

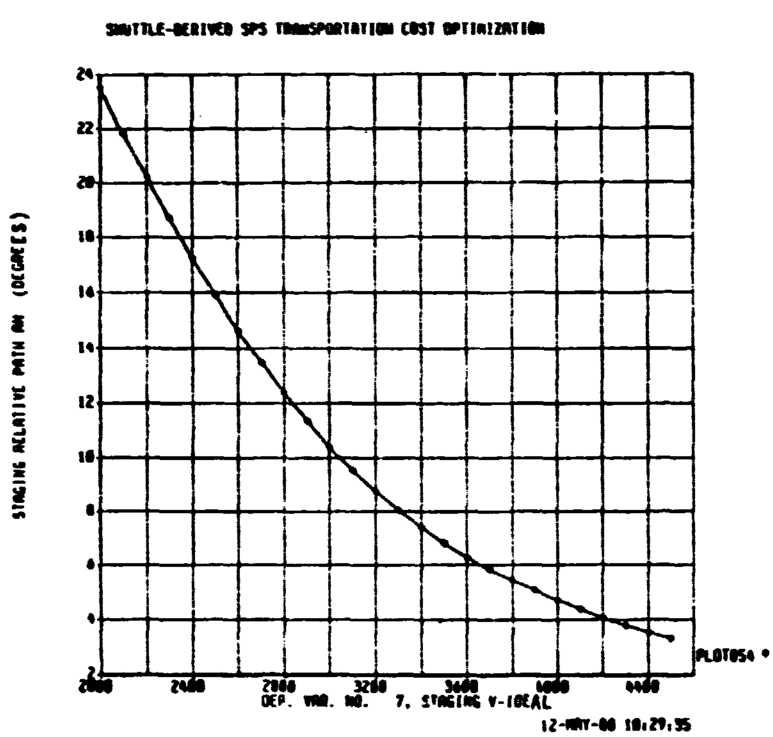

Figure 3.2.5. Model Inputs (Continued)

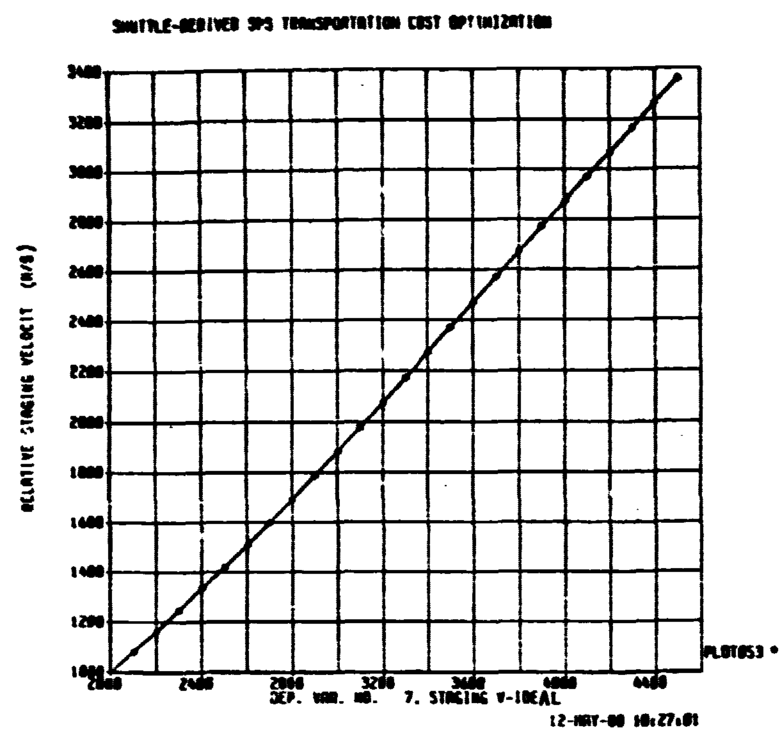

Figure 3.2-6. Model Inputs (Continued)

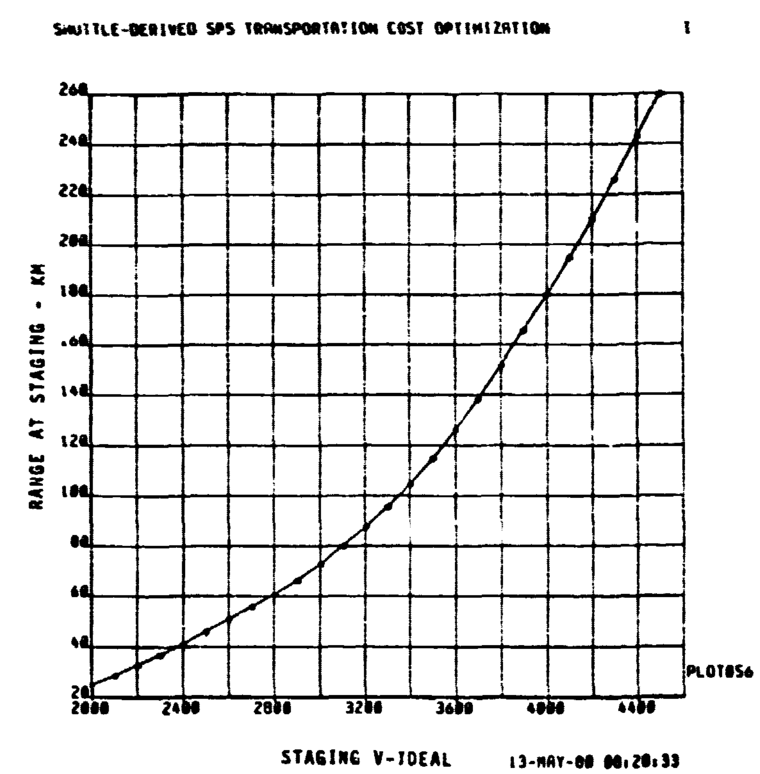

Figure 3.2-7. Model Inputs (Continued)

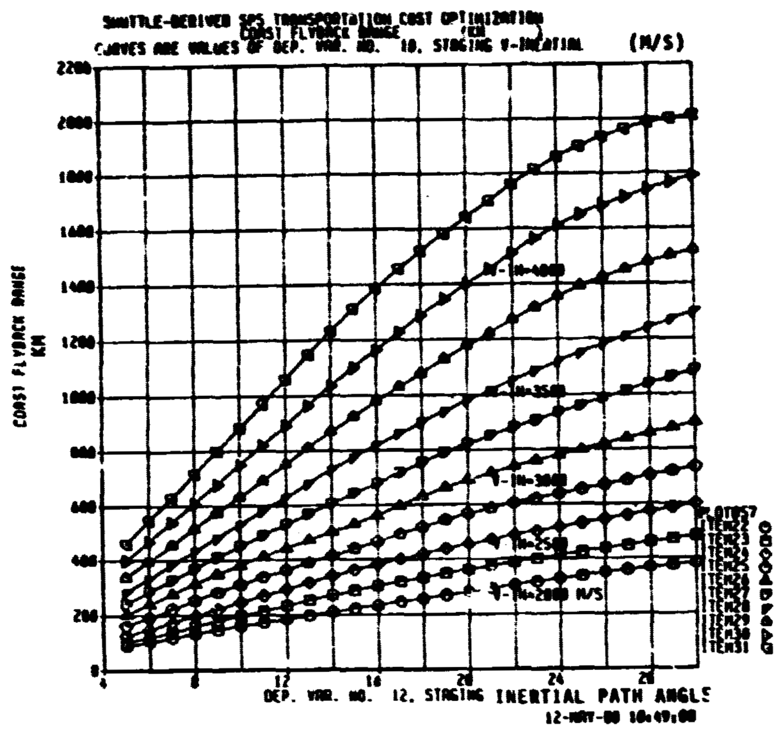

Figure 3.2-8. Model Inputs (Continued)

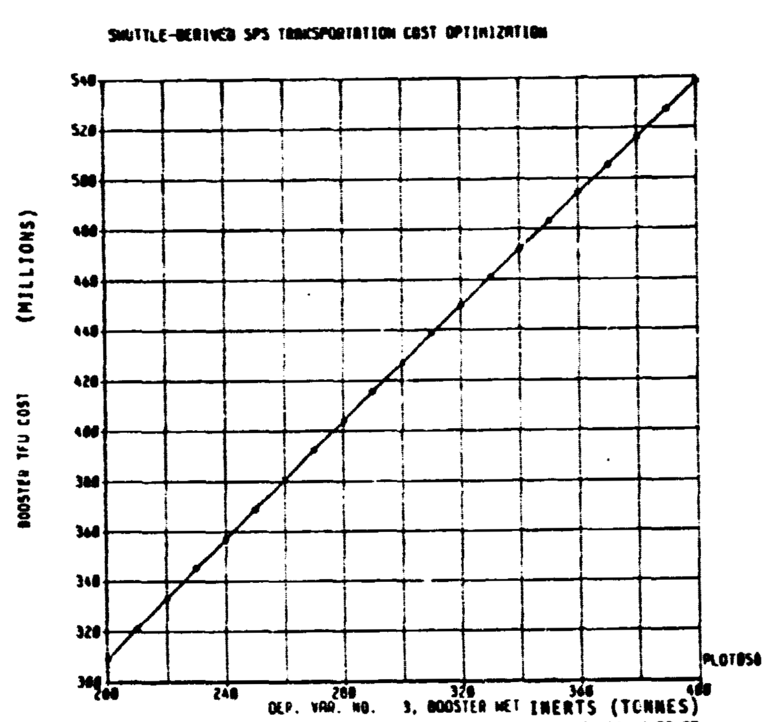

Figure 3.2-9. Model Inputs (Continued)

ET costs were computed based on the theoretical first unit for the basic ET and on a delta theoretical first unit for the additional mass of payload bay which in turn depends upon the payload deliverable per flight. The delta TFU is shown in Figure 3.2-10.

The propellant transferrable is dependent upon the propellant remaining at staging. For relatively low values of propellant remaining, very little propellant is transferrable since most of it will be vaporized by the tank vapor residuals and the tank wall mass. The model relationship is shown in Figure 3.2-11.

The next several figures summarize results.

The first run of the model examined the importance of booster wing area. Wing area was found not to be a very important parameter as shown in Figure 3.2-13.

Large wing areas actually reduce booster start flyback inerts as the improvement of L/D is more important than the increase in wing mass. This is shown in Figure 3.2-12.

The orbit transfer propellant-to-cargo is the kilograms of propellant per kg of orbit transfer cargo. It improves with greater numbers of ET's but degrades with larger boosters because the ET mass grows with increased cargo capacity. The trend is shown in Figure 3.2-14. Figure 3.2-15 shows the variation in annual numbers of orbit transfer flights. Figure 3.2-16 shows the variation in orbiter personnel capacity for orbit transfer; as expected, the trend is opposite to the numbers of flights.

Figure 3.2-17 shows the annual number of propellant launches. This is driven by the propellant transferable and is a primary cost driver.

Displayed in Figure 3.2-18 is the total annual cost for construction of two SPS's per year as a function of booster propellant load and number of ET's per orbit transfer. It is evident that large boosters are important and that using at least six ET's per orbit transfer is desirable.

The same results are displayed in Figure 3.2-19 in terms of cost per kilogram.

The previous case was rerun for larger booster propellant loads showing some additional reduction in total annual cost up to 6,000 ton boosters as illustrated in Figure 3.2-20. The total annual cost here is about twice that for the small HLLV whereas the booster size is approaching the booster for the large HLLV which had a propellant load of about 7,000 metric tons.

3.3 CONCLUSIONS

A number of developmental requirements are necessary in order to implement the shuttle derived system. These are summarized in Table 3.3-1. Several changes to the external tank are required and orbiter crew accommodations of up to 30-40 crew are needed for the orbit transfer. These crew accommodations can be provided in the payload bay. A new large booster is required and the orbiter/external tank flight operations technology involved in transferring propellant and flying LEO to GEO orbit transfers must also be developed.

The most significant results relate to cost. The recurring cost for the shuttle-derived system is estimated as about twice that of the small heavy lift launch vehicle and the DDT&E, including the large booster and the ET mods is estimated at 60 to 70 percent of the small heavy lift launch vehicle. The shuttle derived system optimizes with payload to

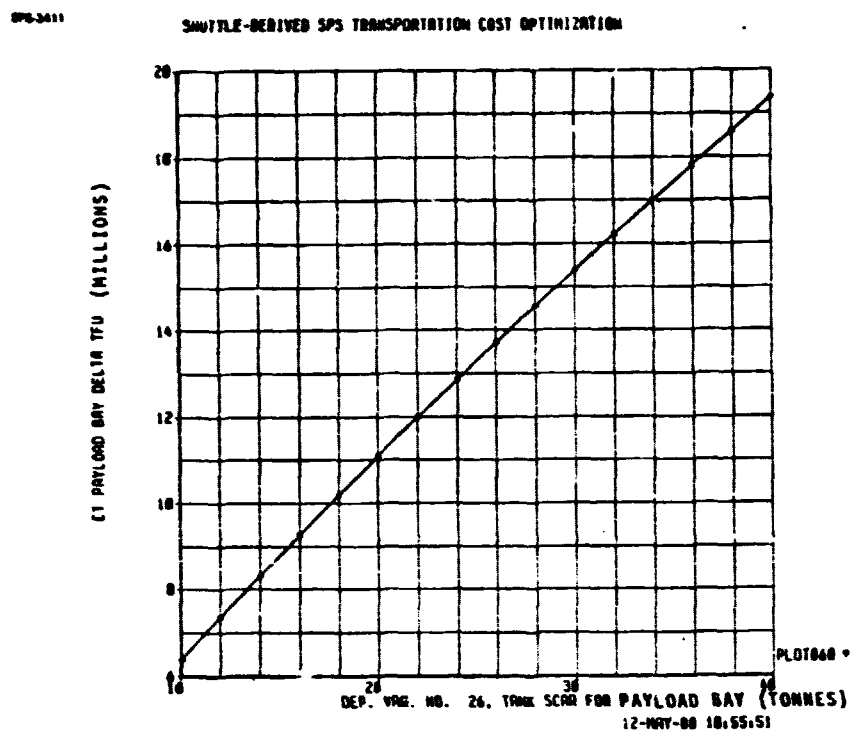

Figure 3.2-10. Mcdel Inputs (Continued)

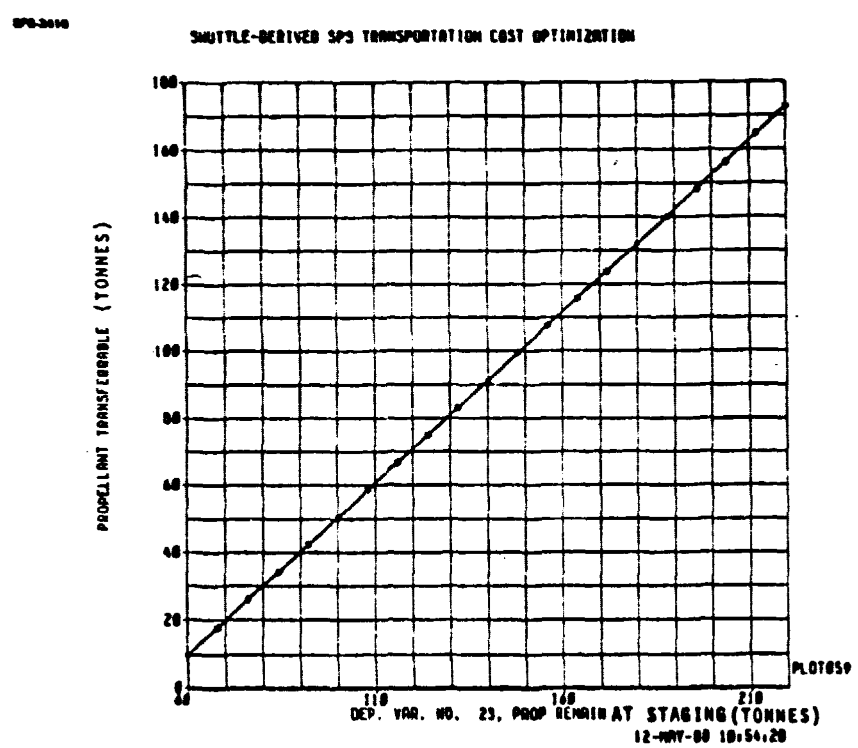

Figure 3.2-11. Model Inputs (Continued)

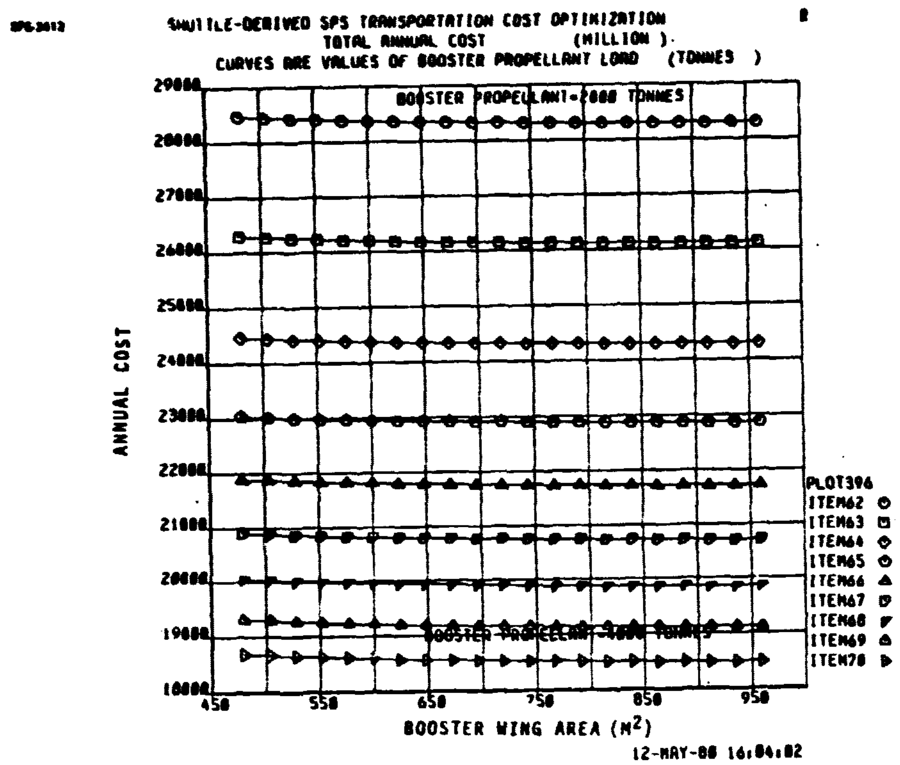

Figure 3.2-12. Wing Area Effects

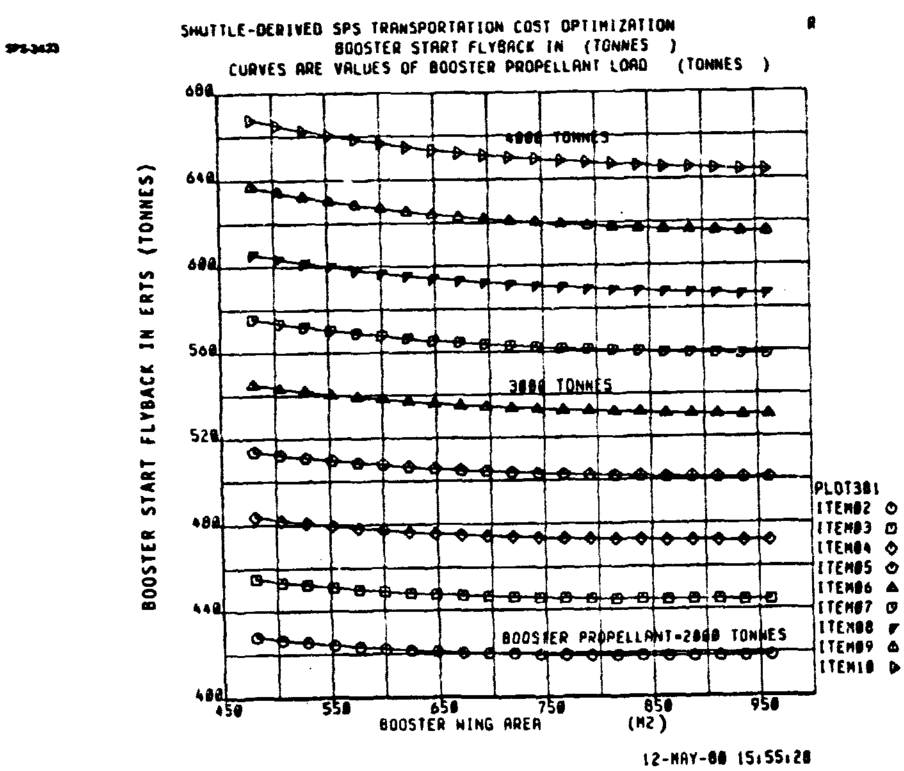

Figure 3.2-13. Booster Start Flyback Inerts

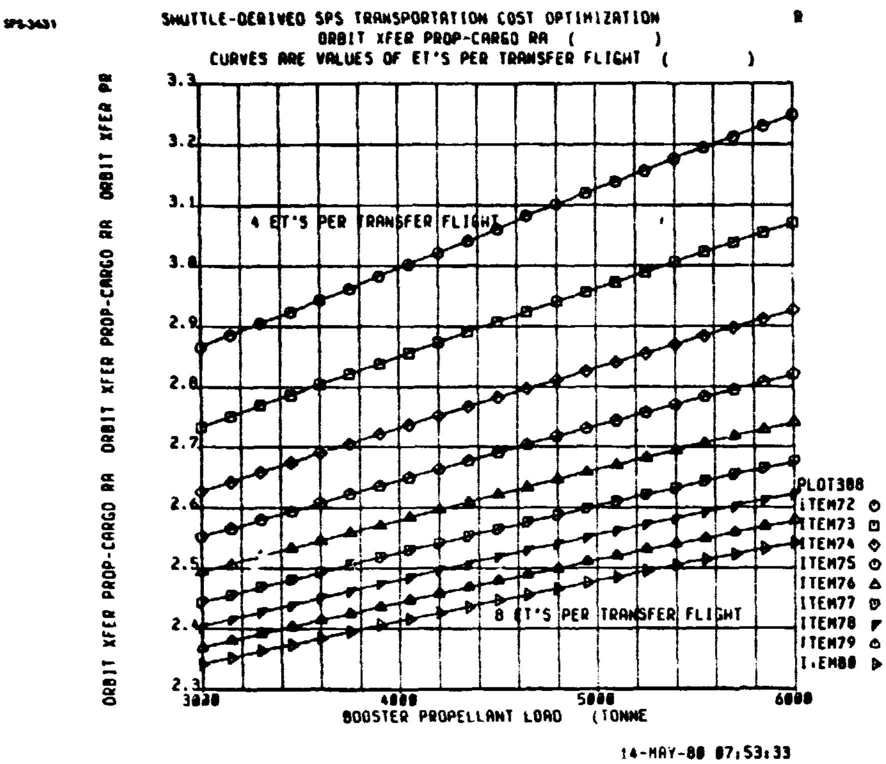

Figure 3.2-14. Shuttle-Derived SPS Transportation Cost Optimization Orbit Transfer Prop-Cargo RA ( )

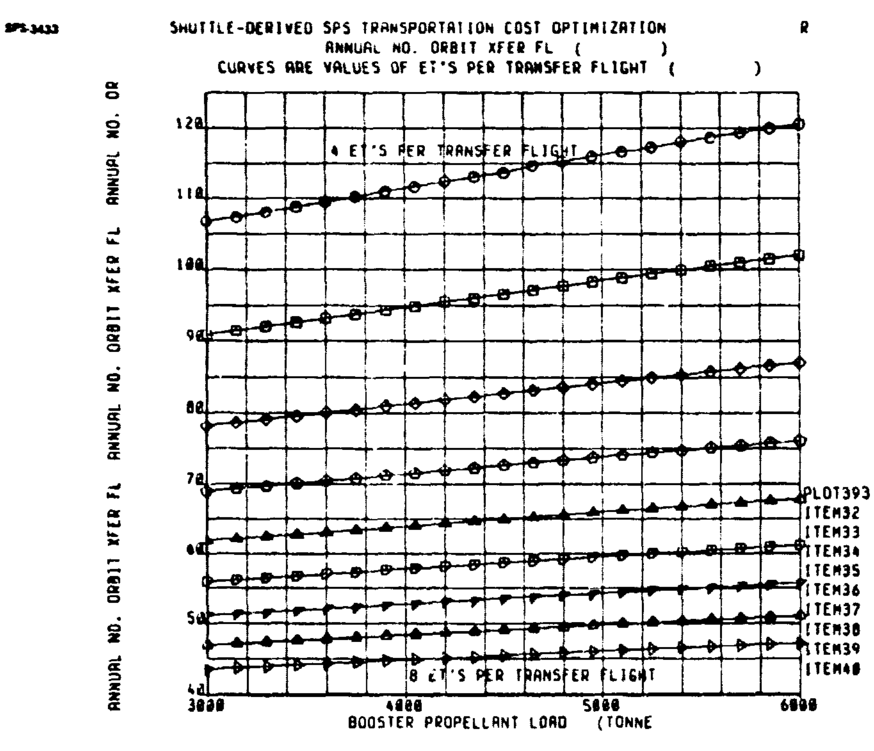

Figure 3.2-15. Shuttle-Derived SPS Transportation Cost Optimization Annual No. Orbit Transfer FL ( )

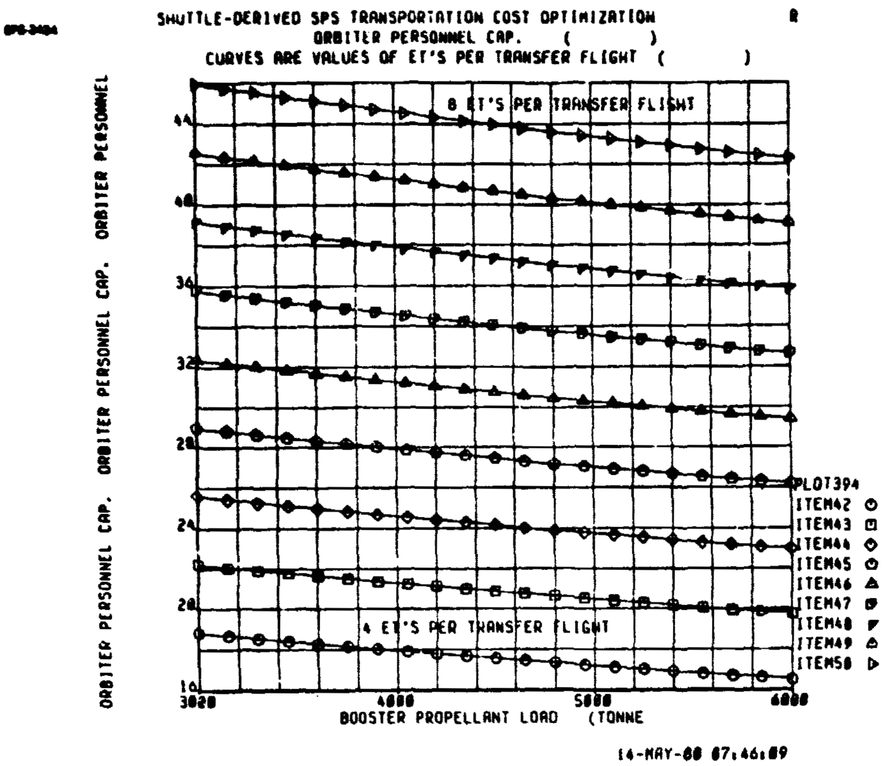

Figure 3.2-16. Shuttle-Derived SPS Transportation Cost Optimization Orbiter Personnel Cap. ( )

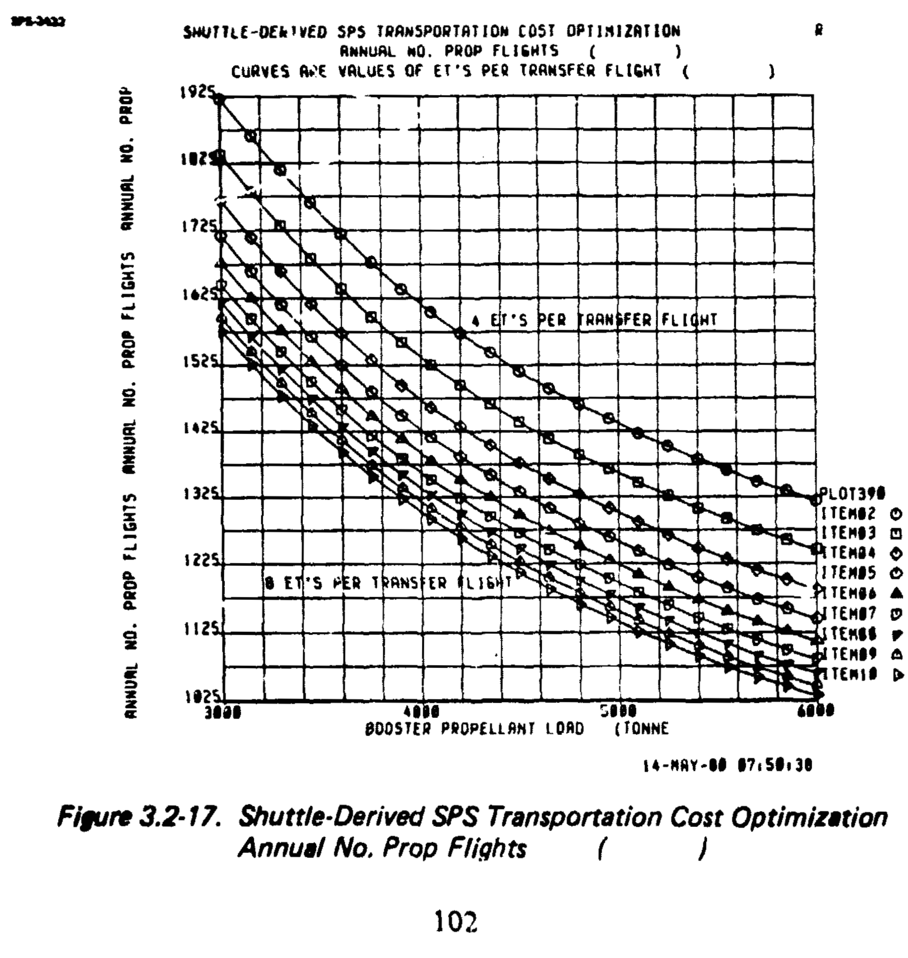

Figure 3.2-17. Shuttle-Derived SPS Transportation Cost Optimization Annual No. Prop Flights ( )

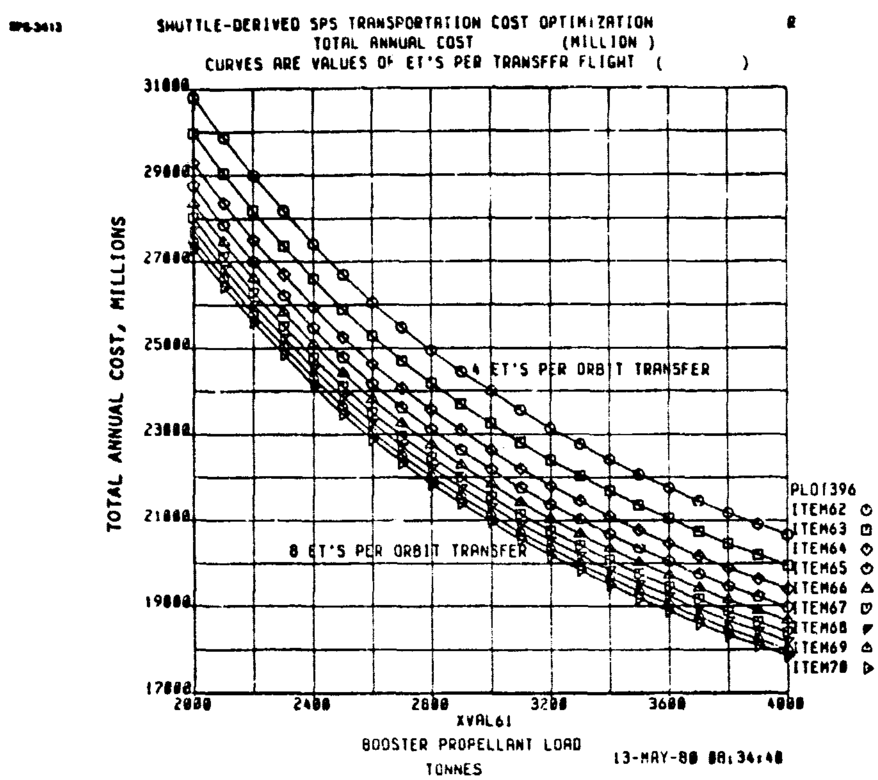

Figure 3.2.18. Shuttle-Derived SPS Transportation Cost Optimization Total Annual Cost (Million)

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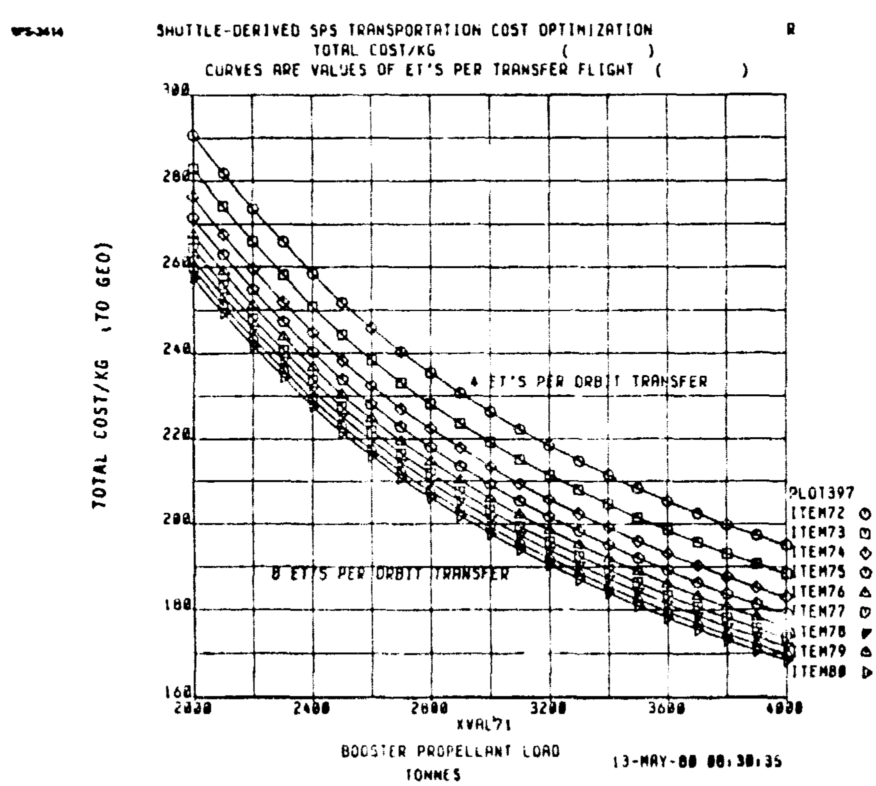

Figure 3.2-19. Shuttle-Derived SPS Transportation Cost Optimization Total Cost/KG ( )

SPS-3438

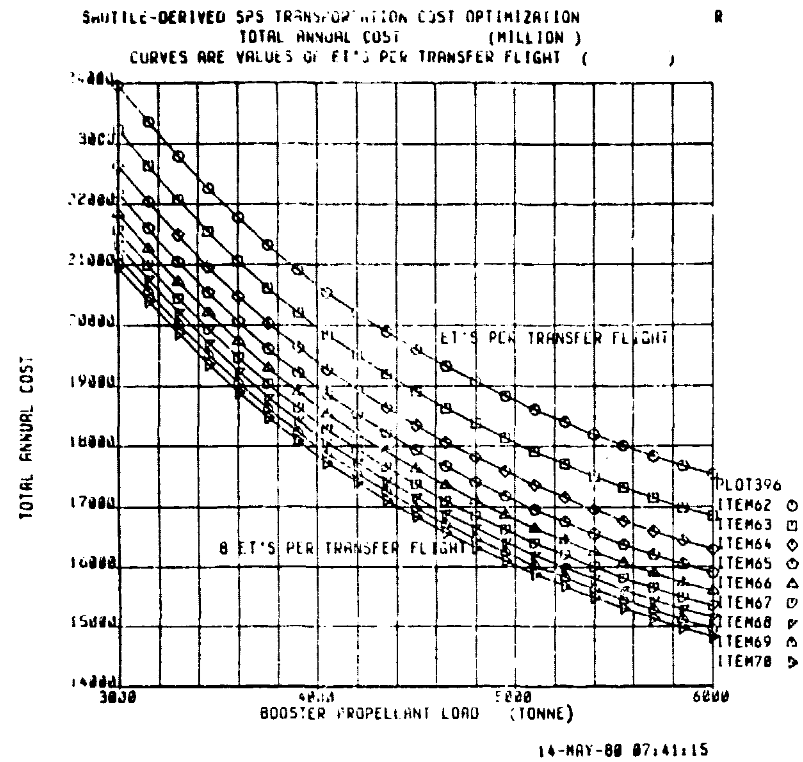

Figure 3.2-20. Shuttle-Derived SPS Transportation Cost Optimization Total Annual Cost (Million)

Table 3.3-1. Shuttle-Derived Development Requirements

SPS-3419

- ET CARGO BAY (CARGO ET'S ONLY)
- ET IMPROVED INSULATION (ORBIT TRANSFER ET'S)
- ET DOCKING (ORBIT TRANSFER ET'S)
- ET PROPELLANT TRANSFER EQUIPMENT
  (PROPELLANT ET'S AND
  ORBIT TRANSFER ET'S)
- ORBITER CREW ACCOMMODATIONS ≈ 30 TO 40
- NEW BOOSTER 5000 TO 6000 TONNES GROSS BOOSTER MASS
- ORBITER/ET FLIGHT OPERATIONS

orbit per flight in the range of 300 tonnes. This payload capacity is too large for many other applications, a criticism also directed at the large SPS reference heavy lift launch vehicle.

It is recommended that the small heavy lift launch vehicle be selected as the SPS reference system. That small vehicle was described in the previous report section. The shuttle derived concept, however, should be retained as an option for further consideration and reexamined in light of shuttle operating experience after a few shuttle flights have been accomplished.

4.0 ELECTRIC ORBIT TRANSFER VEHICLE (EOTV) ANALYSES

4.1 INTRODUCTION

The electric orbit transfer vehicle analysis conducted sensitivity studies relative to the reference EOTV system. The principal subjects of investigation were thermal effects in low Earth orbits and the sensitivity of the vehicle system design to the success of solar array annealing technology.

4.2 THERMAL EFFECTS

The original analyses of the electric orbit transfer vehicle presumed that the solar array output would be equivalent to that expected at geosynchronous orbit without significant thermal radiation effects due to the proximity of the Earth. Much of the orbit transfer propulsion operations, however, take place near the Earth where reflected solar radiation and infrared radiation from the Earth raise the solar array temperature from the geosynchronous orbit value of 40°C to as much as 70°C. The result is a reduction in output from the solar array. Silicon solar cells have a temperature coefficient of approximately 0.4% per degree C. Thus, a 20°C increase in temperature reduces the output by about 8%.

Unlike power supplies for satellites where the supply output must always exceed the demand from the satellite, an electric orbit transfer vehicle may be designed to utilize whatever power output is available from the array. Consequently, in order to investigate the significance of thermal effects, it was necessary to develop a simulation which determined the output of the array as a function of orbit geometry and then applied this output to thrust generation to simulate the orbit transfer mission with thermal effects. In order to do this, thermal analyses were conducted to predict solar array temperature as a function of orbit altitude and aspect angle. Results of these simulations are presented in Figure 4.2-1. These results were incorporated into a table look-up that was made a part of the orbit transfer simulation routine. The orbit transfer simulation was then used to predict orbit transfer performance with thermal effects included.

A second concern is the question of start-up time for the electric thrusters. Once the EOTV emerges from the Earth's shadow, the solar array temperature must stabilize and the electric thrusters must be started before electric propulsion for raising of the orbit can commence. Estimates of the time required to start electric thrusters span a wide range. The most reasonable estimates appear to be a time delay of approximately 10 minutes. This is also consistent with the time required to stabilize the solar array temperature after emergence from shadow. Therefore, a time delay of 10 minutes was examined in the orbit transfer simulations to ascertain sensitivity of orbit transfer performance to time delay. Figure 4.2-2 compares the orbit transfer performance with no time delay or thermal effects to performance with thermal effects only, and to performance with thermal and time delay effects.

The range of solar array temperatures results from changes in orbit aspect. Every 400th integration step is plotted; roughly every five revolutions of the Earth. For this reason the temperature data look like a random sampling. Darkside temperatures were not included in the table lookup as the electric thrust is "shut off" on the dark side by an occultation subroutine included in the simulation.

Chemical thrusters are used in the dark side to maintain attitude control. The thrust is just sufficient to counter gravity gradients; the orbit is not raised by the chemical thrusting. This non-impulse propellant flow reduces the effective specific impulse of the

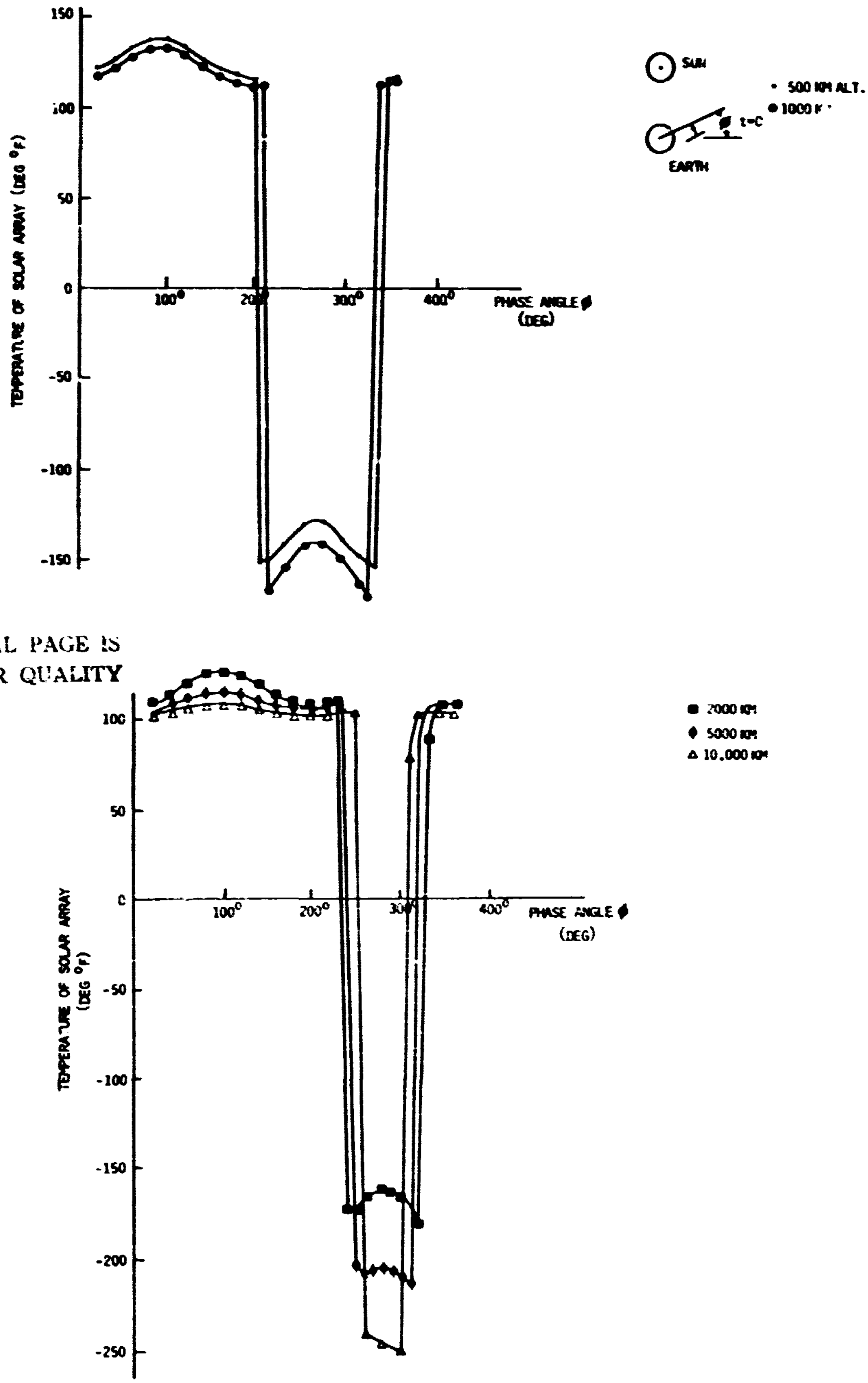

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Figure 4.2-1. Thermal Effects on Solar Array

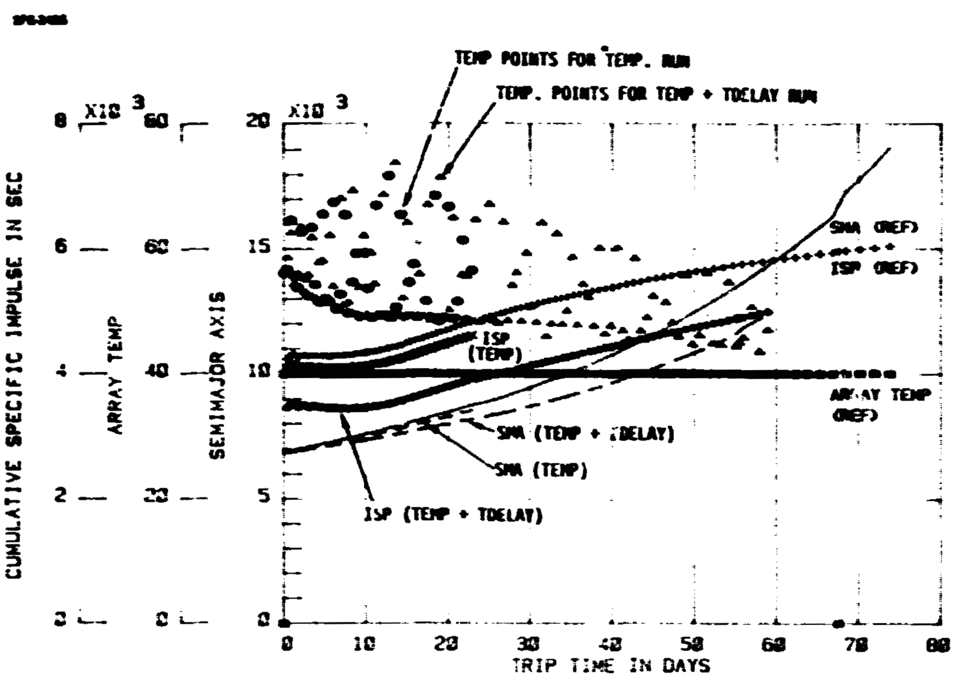


Figure 4.2-2. Orbit Transfer Simulations—Electric Orbit Transfer Vehicle

transfer. As the orbit altitude increases, the cumulative average specific impulse increases because shadowing and gravity gradients both decrease.

Thermal effects slightly increase the transfer time and slightly decrease specific impulse (the latter because the delivered electric impulse is slightly decreased while the chemical is not).

The ten-minute time delay is much more significant as may be seen from the figure. In this case, the chemical impulse delivered is increased at the same time the electric is decreased.

These degradation effects may be expressed in terms of a correction factor that corrects the trip time performance of the system for reduced output due to thermal effects and increased trip time due to start-up delay effects. The estimated Isp correction factor derived from these results was 0.785 (The actual Isp is 0.785 of the electric Isp considering thermal and time delay effects). Similarly, the actual trip time is extended about 35% from the idealized, unocculted case. The systems analysis results employed correction factors representing the combined effects of thermal and time delay.

4.3 MAGNETOSPHERE ALTERATIONS

Further speculation has been directed to the question of disruption or alteration of the Earth's magnetosphere by the high-power electric propulsion plumes. It is not presently known if this is a significant problem. If it is, substantial mitigation of the problem should be available through use of hydrogen in place of argon as an electric propulsion propellant and use of either arc jet or magnetoplasmadynamic (MPD) thrusters rather than ion thrusters. The reasons are that hydrogen, unlike argon, is quite plentiful in the magnetosphere and further, that the arc jet or MPD thrusters will produce a plasma relatively little ionized compared to that expelled by argon ion engines. MPD thrusters are expected to exhibit somewhat better efficiency at low specific impulse and poorer efficiency at high specific impulse compared to ion thrusters. Figure 4.3-1 shows a projection of MPD thruster performance. It may be compared with the ion engine thruster performance estimate shown in Figure 4.4-6.

4.4 PERFORMANCE UPDATE

Systems analysis of the electric orbit transfer vehicle was conducted employing the ISAIAH computer program routine for operation of an EOTV systems model. The performance segment of the model was based on the generalized trip time equation discussed in Appendix A. This trip time equation allows analysis of orbit transfer, performance, mass, and cost based on closed form expressions employing iteration of electric orbit transfer vehicle mass properties.

The most important part of the simulation is the transfer performance simulation. This is diagrammed in Figure 4.4-1. The critical part of this computation network is the determination of required jet power. The one-way mass ratio is computed from the electric specific impulse, a specific impulse degradation factor determined by 6^{0} of freedom orbit transfer simulations that includes chemical attitude control in Earth's shadow, and the one-way delta v. The electric propulsion power system is sized for the available electric power at the beginning of the up trip. During the up trip, the available electric power will be degraded as a result of passage of the vehicle through the Van Allen radiation belts. Consequently, the trip time expression is divided into an expression

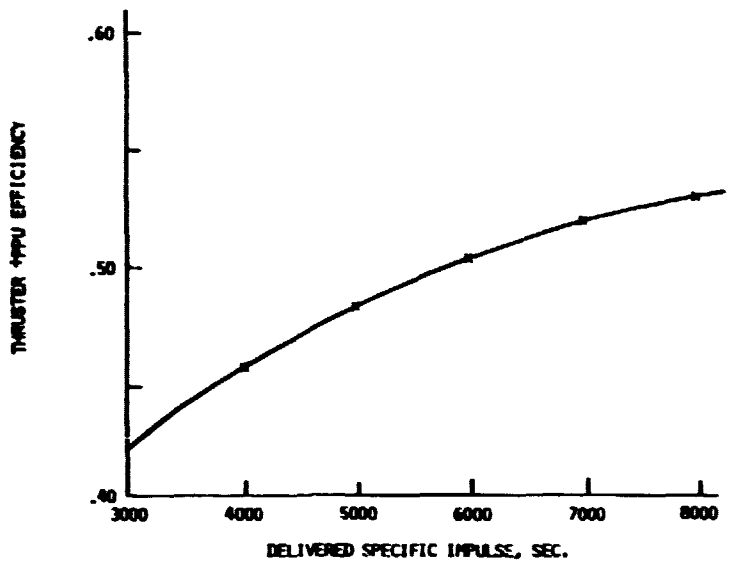


Figure 4.3-1. MPD Performance Projection

SPS-3440

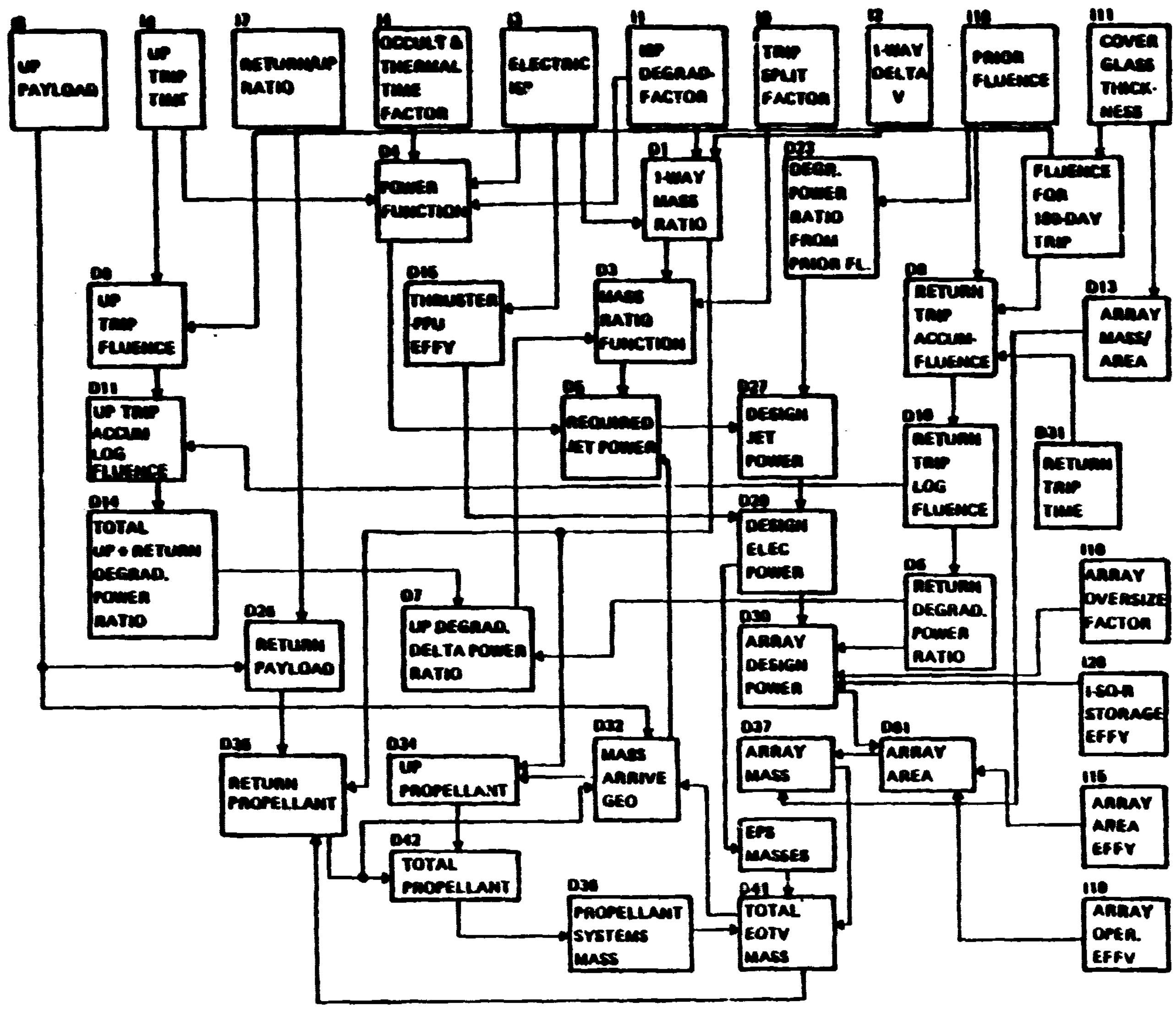


Figure 4.4-1: EOTV Performance Model

relating that portion of the trip time that occurs prior to degradation and a second portion that relates trip time subsequent to the radiation degradation effects. These segments of the trip time expression are incorporated in a mass ratio function.

A second important function is a power function dependent upon the required up trip time, the time factors related to occulatation and thermal effects, and the electric specific impulse and its degradation factor. The power function and the mass ratio function are multiplied together with the electric orbit transfer vehicle mass, arriving at geosynchronous orbit in order to determine the required jet power. This required jet power specifies the power required for the nth trip, (the first trip, the second trip, fifth trip, or whatever). The vehicle design jet power is based on the first trip. Consequently, it is related to required jet power through a power ratio derived from prior exposure to radiation degradation and whatever annealing assumptions way be employed. The design jet power is also translated to design electric power based on thruster and power processor efficiency which in turn is based on the electric specific impulse. The design electric power determines the mass of all elements of the electric propulsion system except the solar array. The solar array itself is designed to provide an initial or array design power that is based on no degradation, thus it is larger than the design electric power by an additional degradation power ratio. The array design power determines the array area and the latter then determines the array mass through incorporation of the array mass per unit area, in turn a function of cover glass thickness. These mass estimating factors allow determination of the total EOTV mass which is then fed to calculation of the mass arriving at geosynchronous orbit which in turn is fed back to the required jet power. The iterations implied in this network are handled automatically by the ISAIAH methodology.

Several of the input interrelationships were provided in the form of tables. These are displayed in Figure 4.4-2 through 4.4-10.

4.5 MASS AND COST ESTIMATES

The EOTV mass is calculated from high-level mass estimating factors relating the solar array mass, the power processor mass, thruster mass, and auxiliary propulsion masses to array and design electric powers respectively. These masses are then apportioned to lower level mass estimates as described in Figure 4.5-1. Cost estimating includes consideration of investment cost, HLLV lift cost, and EOTV amortization and trip time costs as diagrammed in Figure 4.5-2.

Four cases were investigated. The first is the reference EOTV case with 75 micron cover glasses, argon ion thrusters with solar array annealing. The cost per kilogram results for this case are illustrated in Figure 4.5-3. The second case examined was the same system without solar array annealing. Array degradation is so rapid that one may expect no more than 3 trips. The system cost with 3 trips was hand calculated as about 80/kg.

Increasing cover glass thickness allows substantially more trips (up to 10). The fluence for 180 day trip as a function of cover glass thickness was presented in Figure 4.4-2. The resulting cost effects are presented in Figure 4.5-4. This clearly shows that thicker coverglasses are preferable to short life.

Figure 4.5-5 shows the expected cost effects of the use of the MPD thrusters with an otherwise reference system. Figures 4.5-6 and -7 compare the cost effects on the thick-cover argon ion system and the MPD system of the thermal and time delay effects predicted from the orbit transfer simulation analysis.

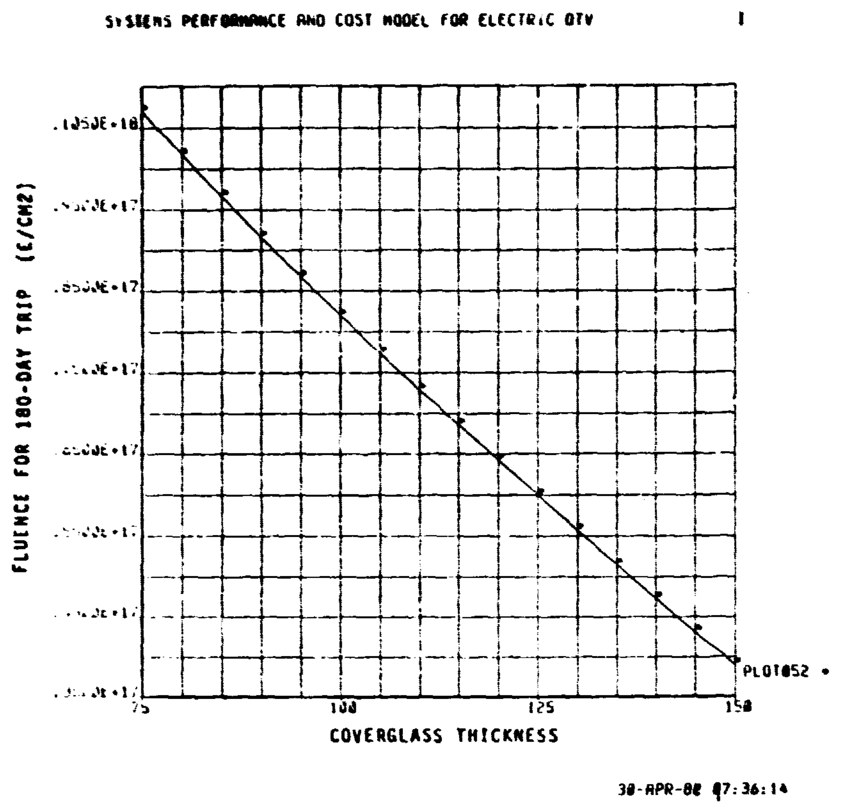


Figure 4.4-2. Systems Performance and Cost Model for Electric OTV

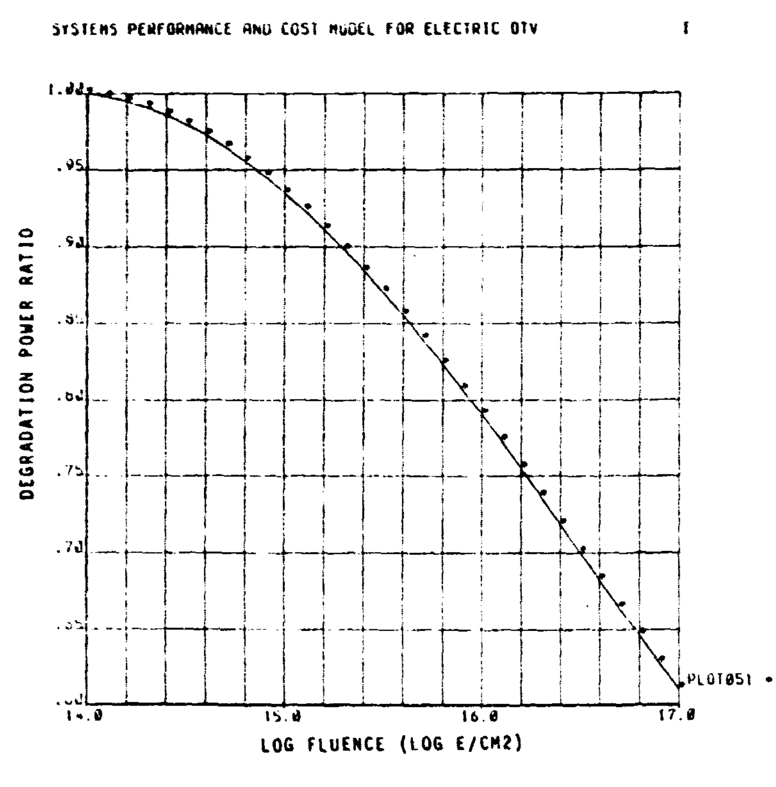


Figure 4.4-3. Systems Performance and Cost Model for Electric OTV

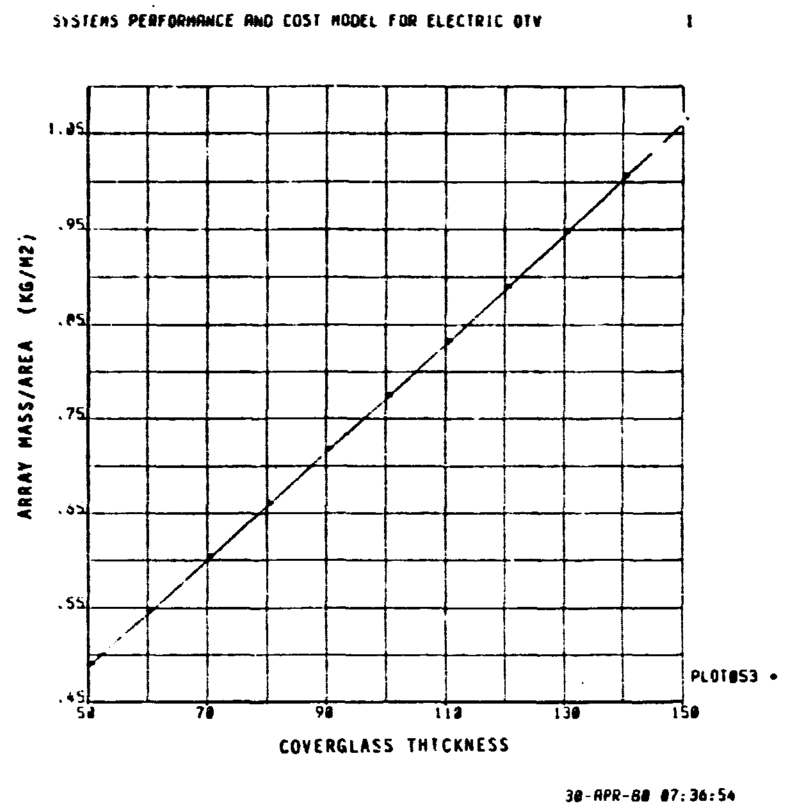


Figure 4.4-4. Systems Performance and Cost Model for Electric OTV

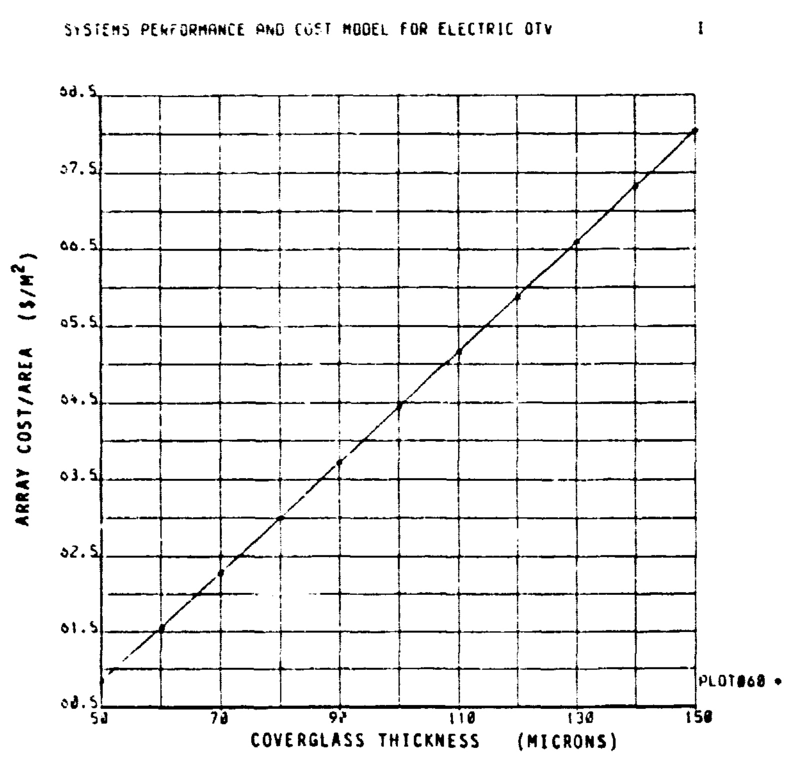


Figure 4.4-5. Systems Performance and Cost Model for Electric OTV

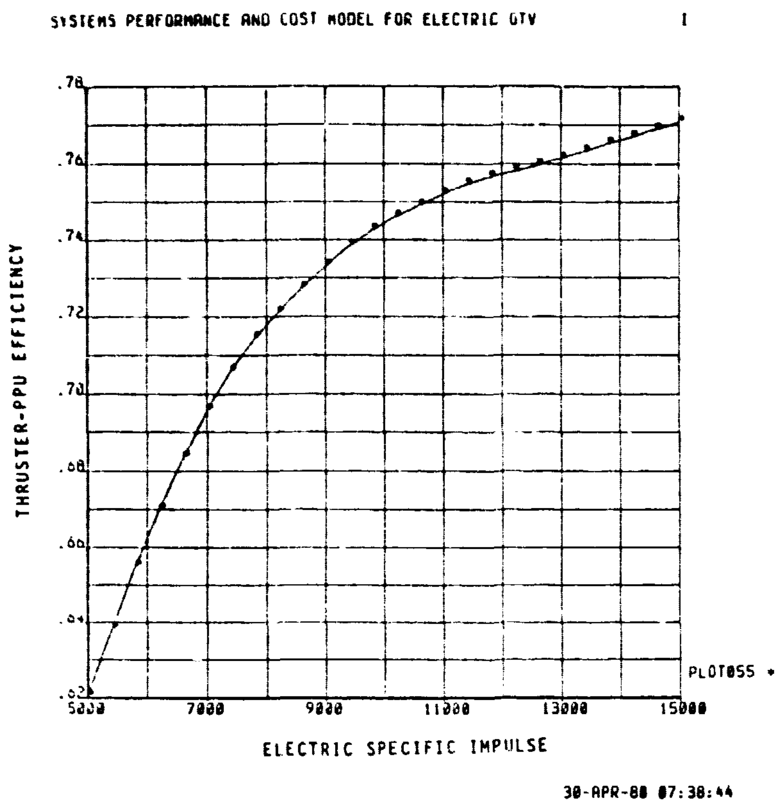


Figure 4.4-6. Systems Performance and Cost Model for Electric OTV

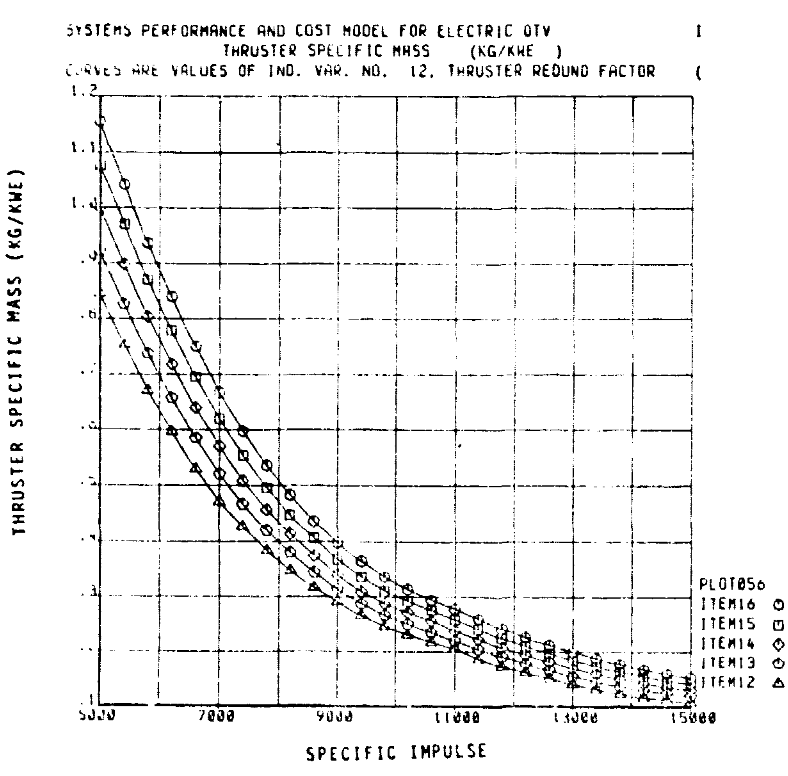


Figure 4.4-7. Systems Performance and Cost Model for Electric OTV

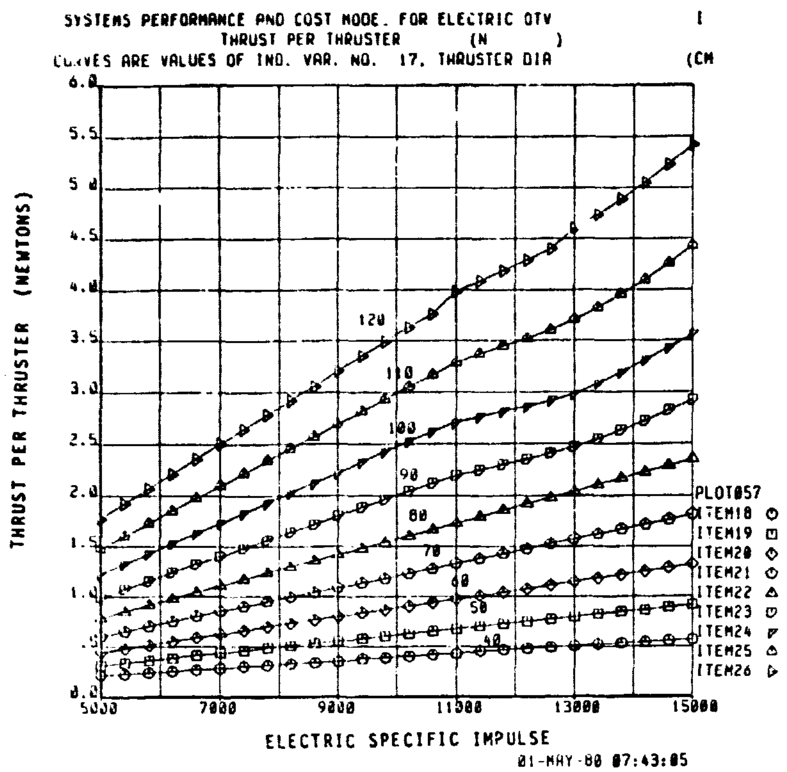


Figure 4.4-8. Systems Performance and Cost Model for Electric OTV

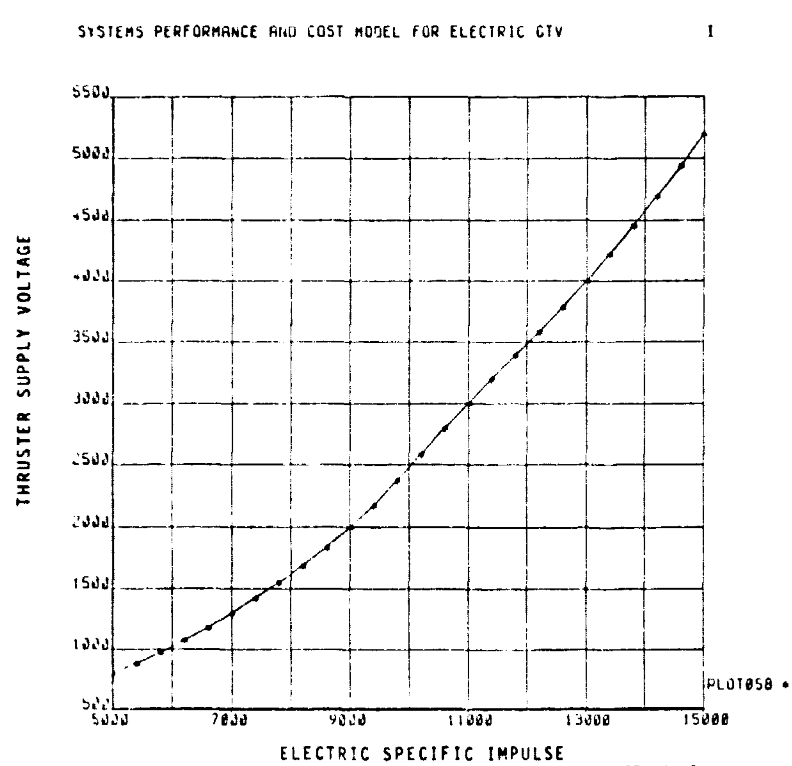


Figure 4.4-9. Systems Performance and Cost Model for Electric OTV

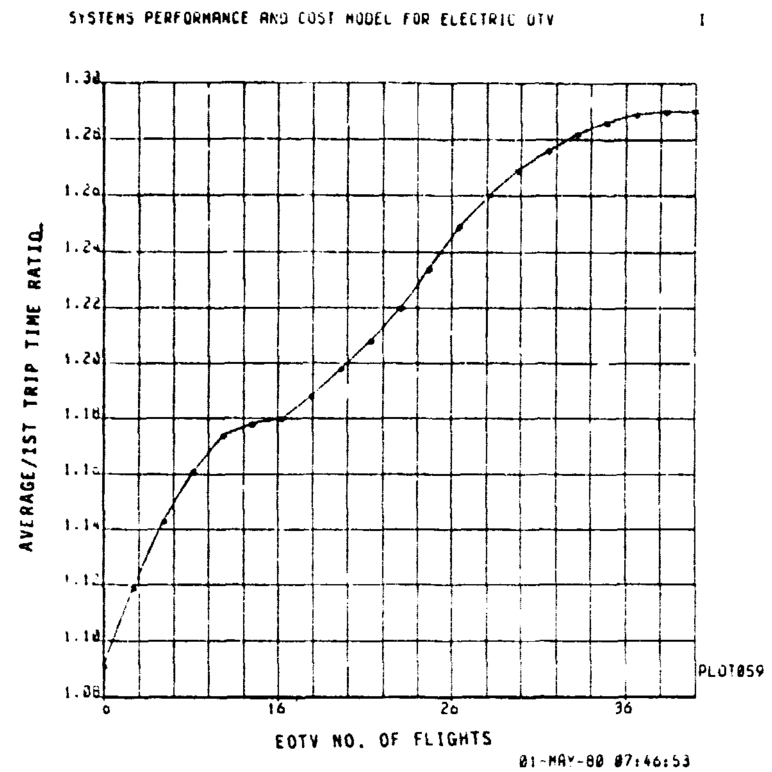


Figure 4.4-10. Systems Performance and Cost Model for Electric OTV

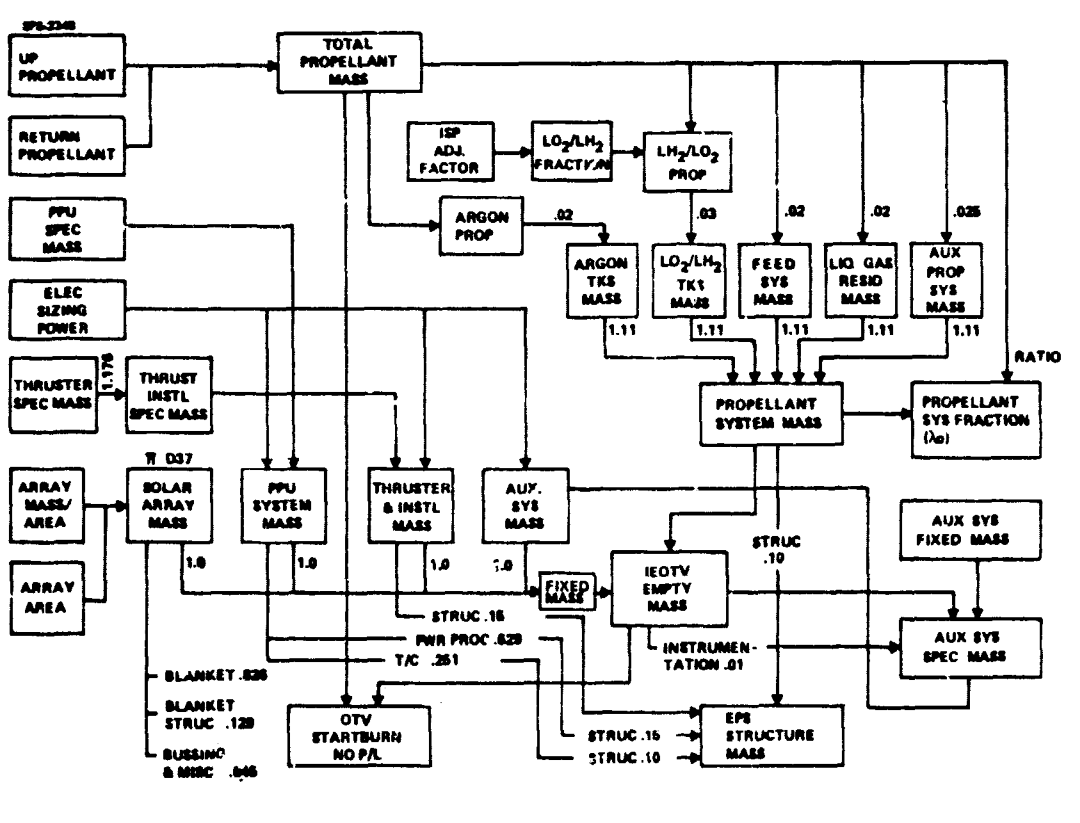


Figure 4.5-1. Mass Estimating Submodel

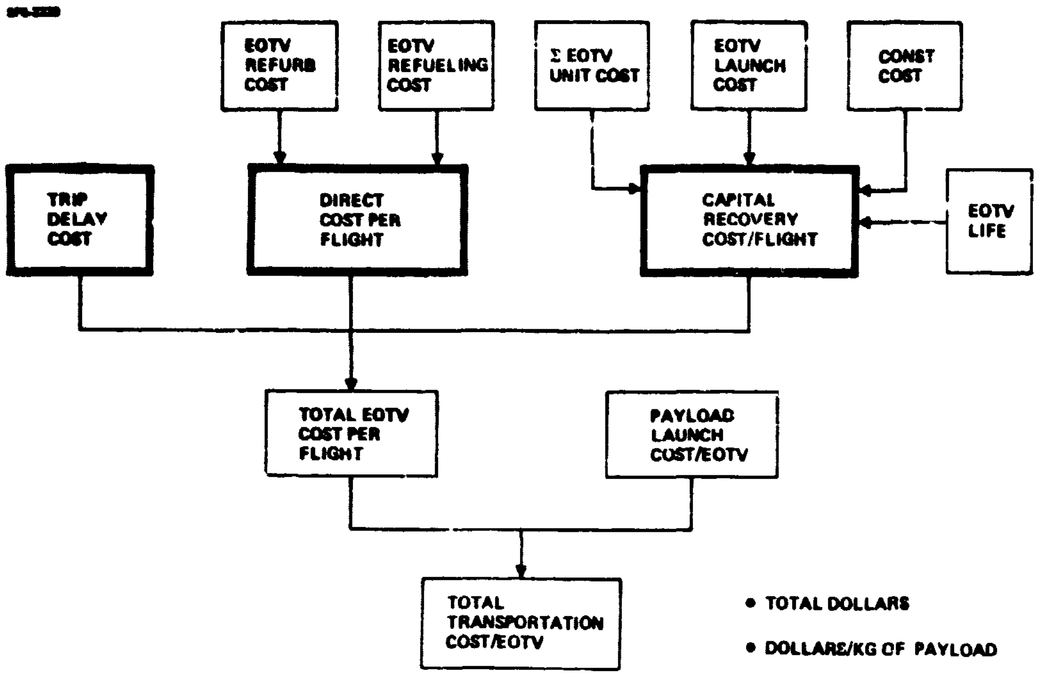


Figure 4.5-2. EOTV Flight Cost Factors

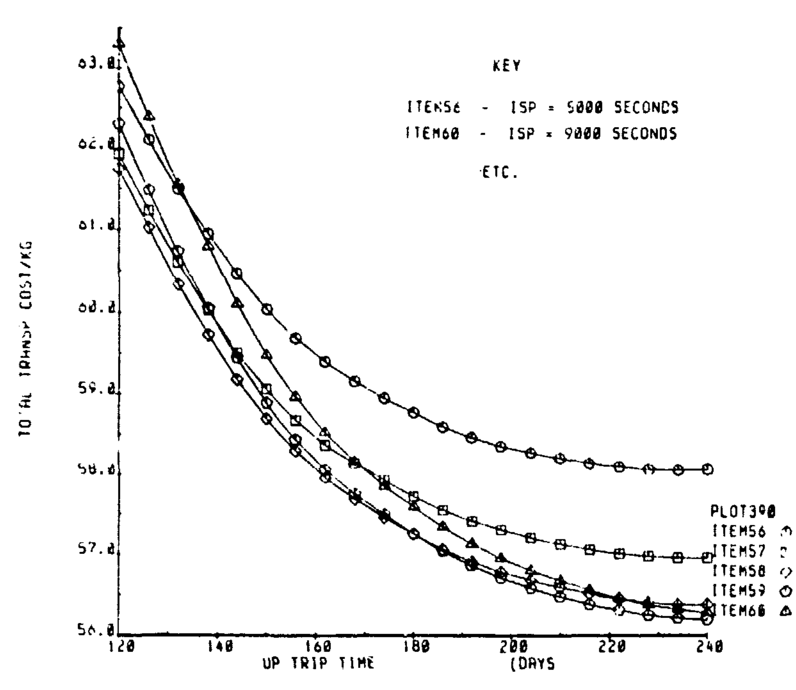


Figure 4.5-3. Systems Performance and Cost Model for Electric OTV

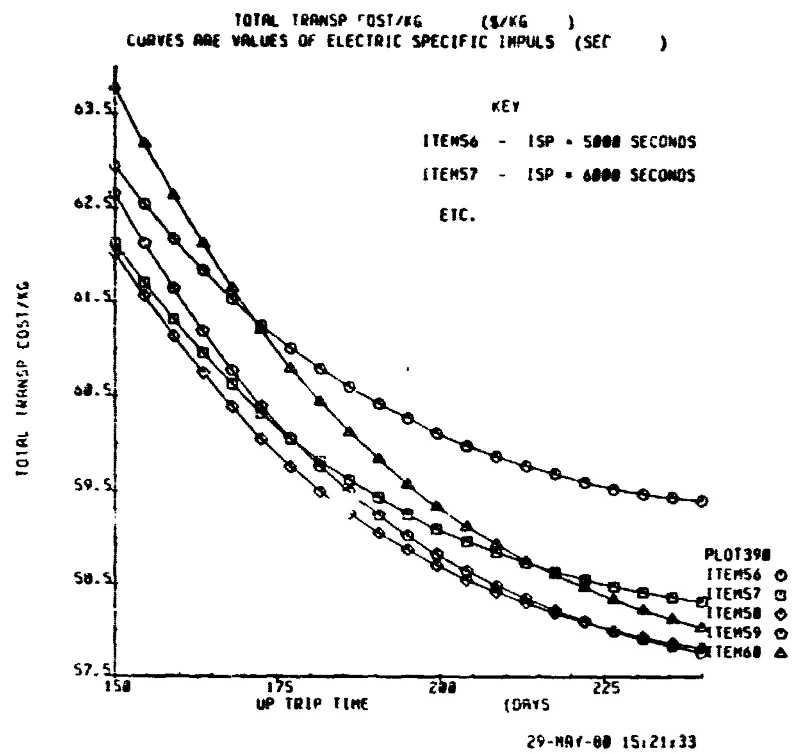


Figure 4.5-4. Systems Performance and Cost Model for Electric OTV

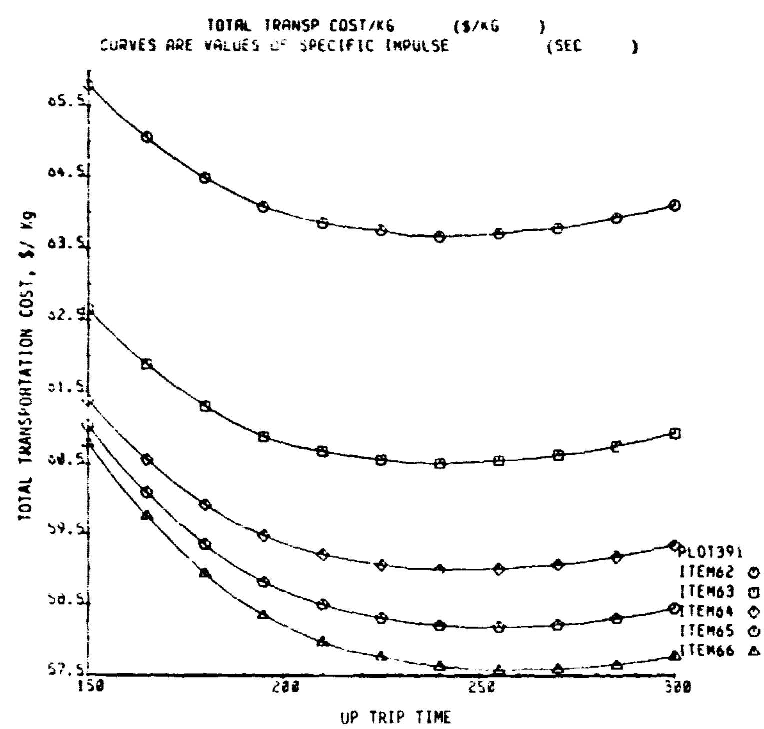


Figure 4.5-5. MPD EOTV: 3 Mil CG. Annealing, No Then ·al/Startup (EOTVA)

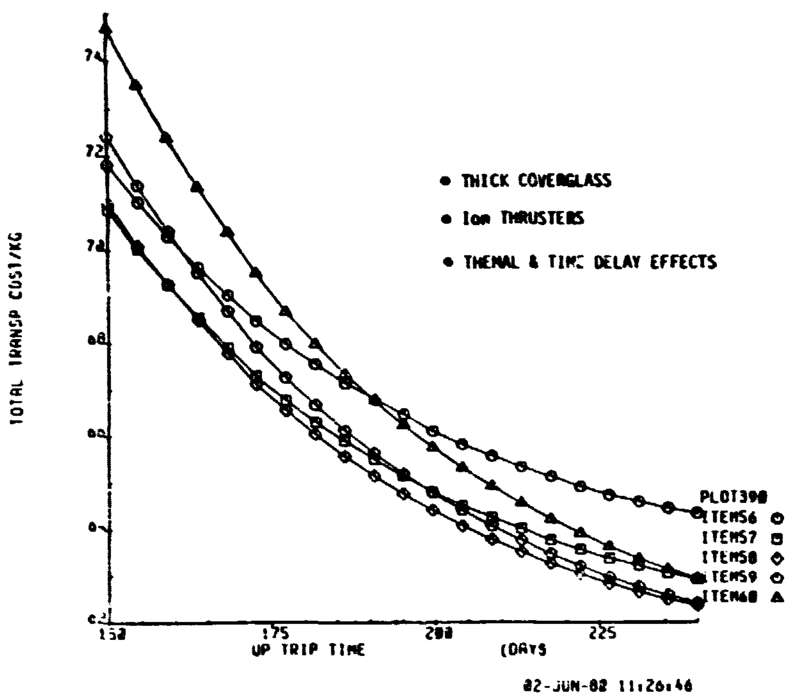


Figure 4.5-6. Systems Performance and Cost Model for Electric OTV

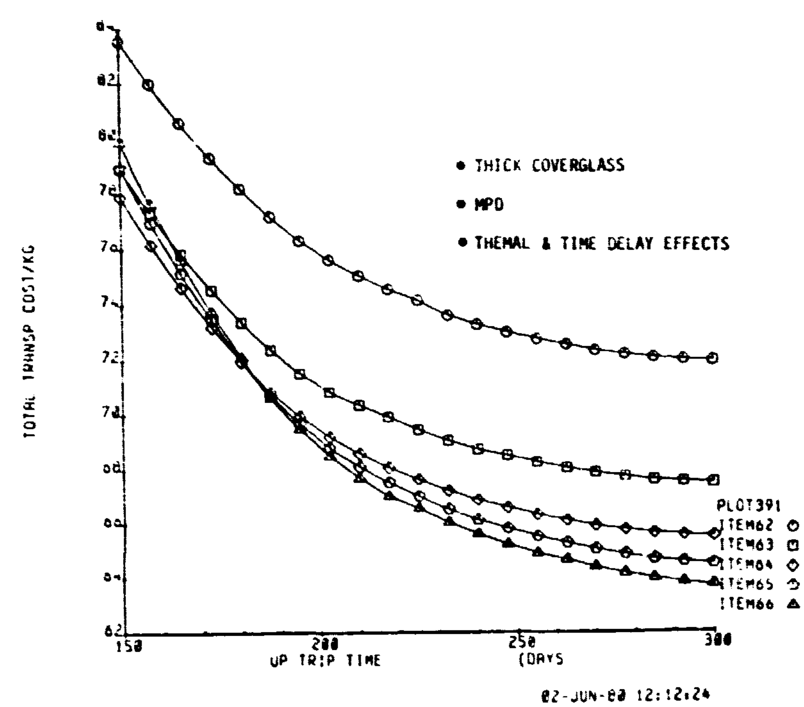


Figure 4.5-7. MPD EOTV (EOTVM)

Finally, Figure 4.5-8 is a bar chart comparing costs of the various systems investigated to a chemical orbit transfer vehicle system cost, all based on the same HLLV launch cost estimates. Results of Figure 4-5-8 were taken for trip times near 180 days. Longer trips are somewhat more cost-effective for the penalized EOTV cases as was shown on the earlier charts.

Additional data and plots from these studies are included in Appendix B.

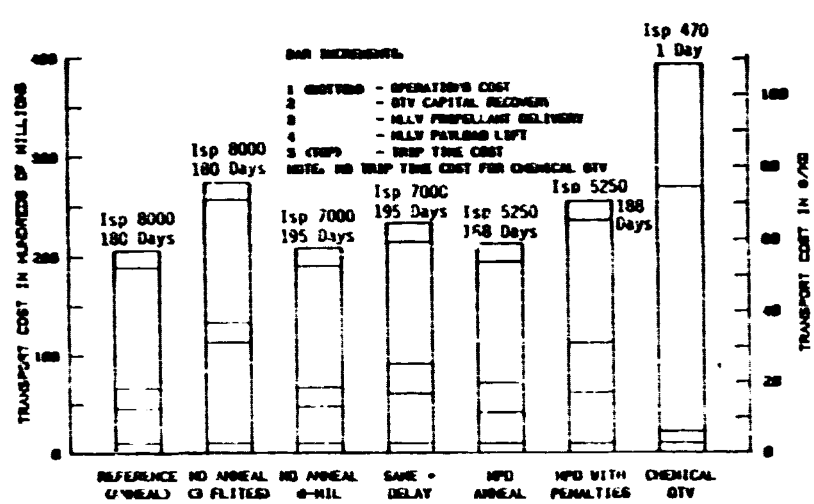


Figure 4.5-8 SPS Transportation Cost Comparison

5.0 TECHNOLOGY

No new technology was developed as a result of this study. Three transportation technology recommendations were developed:

1. Hydrogen MPD arc jets appear to be viable as a backup propulsion mode for electric orbit transfer vehicles, should argon ion engines prove to be environmentally detrimental. This conclusion is based on forecasts of MPD performance developed by Princeton and JPL, with duty cycle assumptions developed by Boeing. EOTV costs are sensitive to specific impulse and efficiency. For the hydrogen MPD thruster to be a viable backup it needs an Isp of at least 5000 seconds and an efficiency of at least 50%. (Present projections exceed these targets). Furtherance of MPD technology to provide a more concrete assessment of capabilities is strongly recommended.

2. The EOTV was found to be very sensitive to electric propulsion start delays. A ten-minute delay (after leaving Earth's shadow) increases LEO-to GEO costs almost 10%. Accordingly, research leading to minimal propulsion startup times is strongly recommended.

3. Further research on solar cell radiation degradation and annealing should be given high priority.

6.0 CONCLUSIONS

1. The shuttle derived transportation system was found to be of sufficient interest to be retained as an option for further consideration. Its launch-to-orbit cost performance approaches that of a more conventional HLLV, but only at large payload capabilities exceeding 250 tonnes. The orbit-to-orbit cost performance is significantly less than that for the EOTV.

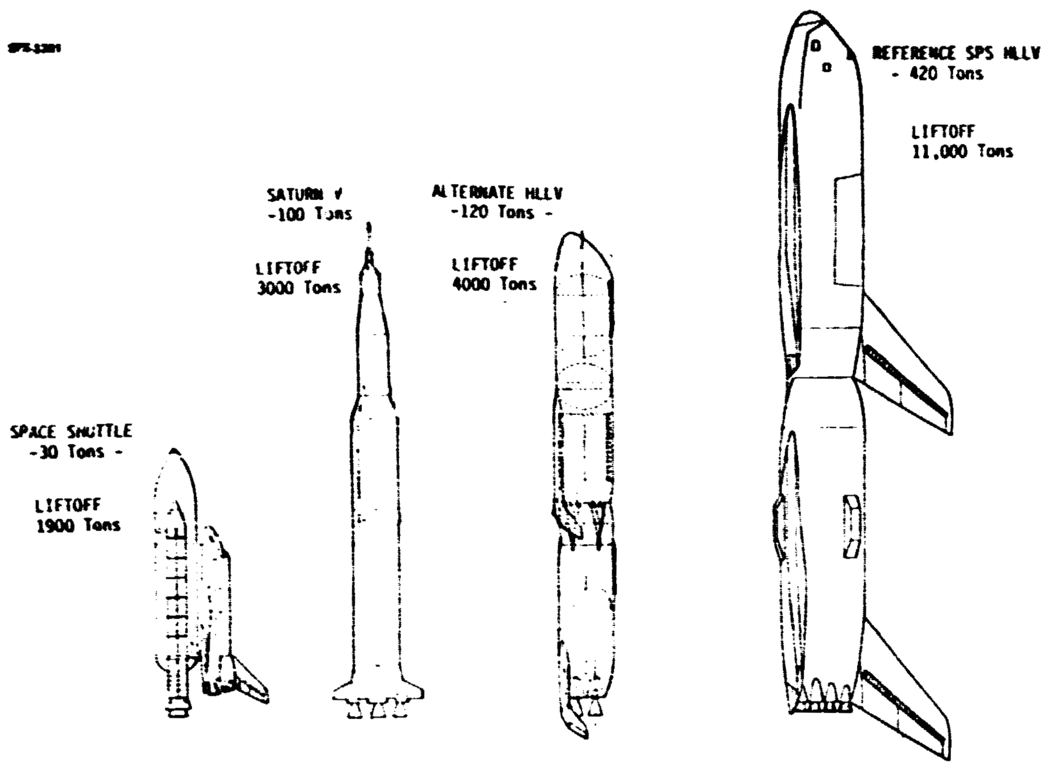
2. A "small" heavy lift launch vehicle was found to be highly attractive for SPS transportation. Significant nonrecurring cost advantages are obtained with only minor recurring cost penalties. The specific vehicle analyzed has about the right characteristics:

Payload Bay Volume	22 x 11 meters cross-section 15 meters long
Lift Capability	125 tonnes to 500 km 30° orbit
Liftoff Mass	4000 tonnes

   This vehicle is compared with others in Figure 6-1.

3. The electric orbit transfer vehicle is a viable option without annealing. If annealing cannot be developed, significantly more shielding (150 microns to 200 microns coverglass) should be used to increase array lifetime. Thermal effects in low Earth orbit are not very important; the effects of electric thrust start delays are more significant. A ten-minute start delay leads to about a 10% cost penalty.

4. Hydrogen MPD arcjets can be used if argon ion engines prove unsuited for EOTV use because of magnetosphere effects. With present estimates of MPD performance, the ion option provides about 10% better cost performance.



Launch Systems Size Comparison

7.0 RECOMMENDATIONS

1. The small HLLV should be adopted as the SPS reference launch vehicle.

2. A study should be performed to assess applicability of this small HLLV to alternate missions in the post-1990 period. The study should attempt to develop an evolution strategy for national heavy-lift transportation capability, including interim systems employing shuttle elements as well as shuttle improvements.

3. The shuttle-derived transportation option should be retained as a backup and examined further after initial shuttle flight experience is obtained.

4. The electric orbit transfer vehicle (EOTV) should be retained as the reference orbit-to-orbit cargo system. Three technology efforts were identified:

    a. Hydrogen MPD arcjets
    b. Rapid startup of electric propulsion
    c. Additional research on solar array radiation degradation and annealing

8.0 REFERENCES

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2. SPS Upper Atmosphere Concerns, Attachment 5 to Monthly Progress Report No. 5, Solar Power Satellite System Definition Study (Contract NASA-15636), November 19, 1978.

3. Telecon with Larry J. Runyan (author of the Sonic Overpressure Analysis in Ref. 1), February 21, 1980.

4. Telecon with Peter Brennan (an associate of the author of the launch noise analysis in Ref. 1), February 25, 1980

5. Telecon with Peter L. Maricich (author of the effluent deposition analysis in Ref. 2), February 25, 1980.

6. Telecon with Calvin Hurd (IUS system safety staff), February 26, 1980.

7. Solar Power Satellite System Definition Study, Phase II, Volume 2, Reference System Description (Contract NAS9-15636). Boeing, D180-25461-2, November 1979.

8. Caluori, V., Comcad, R. T., and Jenkins, J.C., Technology Requirements for Future Earth-to-Geosyncronous Orbit Transportation Systems; NASA Contractor Report 3265 (Contract NAS1-15301), April 1980.

9. Solar Power Satellite System Definition Study, Phase II, Volume 3, Operations and Systems Synthesis (Contract NAS9-15636), Boeing, D180-25461-3, November 1979.

APPENDIX A

ELECTRIC PROPULSION SYSTEMS ANALYSIS USING THE TRIP TIME EQUATION

Presently contemplated applications of electric propulsion include planetary and comet missions and Earth orbital missions. Analysis of the former is complicated by the fact that mission delta V and trip time are interrelated; trajectories must be found and optimized by numerical integration. The mission delta V for Earth orbit missions, however, is essentially independent of trip time. Analyses that are good approximations (within a few percent) are possible using closed-form equations.

Delta V

The mission delta V for coplanar circular orbit transfers, e.g., LEO to GEO, is well-approximated by the Tsieu formula. This states that the low-thrust delta V to change orbits is just the difference in orbit velocities. Example: the orbital velocity at 500 km altitude is 7613 m/s. The velocity at GEO is 3075 m/s. Hence the low-thrust delta V is 4538 m/s

If a plane change is required (as is usual) the delta V calculation is no longer so simple. An optimization is required, because thrust can be used to change plane and altitude at the same time.

Retaining the circular orbit approximation of Tsieu, one can perform an explicit double integral to get delta V to change plane and altitude. An optimal law for plane and altitude change yields about 5850 m/s for orbit transfer from 30^0, 500 km to geosynchronous orbit. 6000 m/s was used for this analysis.

Trip Time Equation

With the delta V for a given mission specified, a very simple equation can be derived to characterize the electric orbit transfer vehicle. It is often called the "trip time equation."

APPENDIX A

It is the low-thrust analog of the Tsiolkovskii equation; it has the same wide applicability. It is very simply derived as follows:

$$t = m_p/\dot{m}_p \text{ (trip time in seconds)}$$

where m_p is propellant mass and $\dot{m}_p$ is mass flow rate.

Jet power is expressed as:

$$P_j = \dot{m}_p u^2/2$$

where u is jet velocity.

$\dot{m}_p$ is therefore expressed as $\dot{m}_p = 2p/u^2$

Substituting in the previous equation,

$$t = (m_p/u^2)(2p_j)$$

Employing now the definition of the terms μ from the Tsiolkovskii equation

$$\left(\mu = \exp \frac{\Delta v}{u}\right) \qquad \mu = \frac{m}{m - m_p}$$

where m is total startburn mass less propellant mass; and solving for m_p;

$$m_p = m(\mu - 1)$$

Substituting in the above,

$$t = m(\mu-1)u^2/(2p_j) = \zeta(\mu-1)u^2/2$$

where ζ is the specific power-to-mass ratio of the *total* inert mass, in kg/watt. If it is desired to show separately the propulsion vehicle and payload mass,

$$t = \zeta'\left(1 + \frac{m_L}{m_e}\right)(\mu-1)u^2/2$$

where t is trip time in seconds,

ζ' is specific mass of the (empty) propulsion vehicle in kg per watt of *jet* power, not including propellant,

m_L/m_e is the ratio of payload to empty vehicle mass

μ is $\exp \Delta v/u$,

u is jet velocity.

APPENDIX A

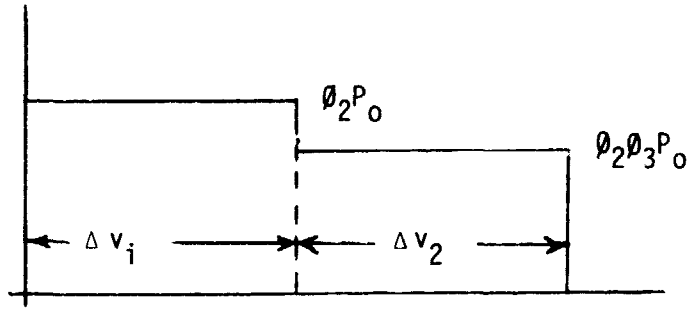
Thus, this one relatively simple equation relates power production performance, payload ratio, mission Δv, jet velocity, and trip time. Note that the equation is in terms of jet power, i.e., $\zeta_j' = \zeta_e'/\eta$ where η is net processor and thruster system efficiency.

The specific mass of the vehicle will vary somewhat with propellant load. It is possible to derive an expanded form of the trip time equation that explicitly includes the dependence of vehicle mass on propelland load. With the ISAIAH methodology, this is not necessary as the propellant system mass can be computed from the propellant mass and fedback to the trip time equation; the ISAIAH iteration procedure closes the loop.

It is important, however, to include other effects: Isp degradation due to Earth shadowing and gravity gradients; trip time extension due to Earth shadowing; and solar array power output degradation due to passage through the Van Allen belts.

The first two of these effects must be assessed by detailed flight simulation. Results of the simulations can, however, be introduced into an electric OTV systems model in the form of "finagle factors." This is done by dividing the up-trip into two parts, a stepwise approximation to radiation degradation.

Power available on up-trip is as sketched:



where the ϕ_2 represents degradation from prior trips, e.g., the down trip, and perhaps earlier flights.

APPENDIX A

The effective Isp, considering chemical thrusting during occulations, is

$$\bar{\bar{I}} = \lambda \bar{I}_e = \frac{f_e(1-\Theta)}{\dot{m}_e(1-\Theta)+\dot{m}_c\Theta}$$

where λ is the correction to electric Isp for chemical thrust, θ is a trip time extension factor, and $\dot{m}_e$ and $\dot{m}_c$ are electric and chemical time-averaged mass flow rates.

This yields

$$\lambda \bar{I}_e = \frac{I_e(1-\Theta)}{1-\Theta+\alpha\Theta}$$

where α is the ratio $\dot{m}_c/\dot{m}_p$

Sloving for α,

$$\alpha = \frac{(1-\lambda)(1-\Theta)}{\lambda\Theta}$$

The time-averaged mass flow is

$$\bar{\dot{m}} = \dot{m}_e(1-\Theta)+\dot{m}_c\Theta$$

$$= \dot{m}_e(1-\Theta+\alpha\Theta)$$

and plugging in for α ,

$$\bar{\dot{m}} = \dot{m}_e\left(1-\Theta+\frac{(1-\Theta)(1-\lambda)}{\lambda\Theta}\,\Theta\right)$$

$$= \dot{m}_e\frac{(1-\Theta)}{\lambda}$$

APPENDIX A

Now we can write the trip time equation in two parts:

$$T_1 = \frac{m_{p1}}{\dot{m}} = \frac{(\mu_a - 1) M_2 u_e^2}{2 p_{j1} \left(\frac{1-\theta}{\lambda}\right)}$$

$$T_2 = \frac{m_{p2}}{\dot{m}} = \frac{(\mu_b - 1) M_3 u_e^2}{2 p_{j2} \left(\frac{1-\theta}{\lambda}\right)}$$

where it is noted that

$$\dot{m}_e = 2 p_j / u_e^2 ; \quad \dot{m} = 2 p_j \left(\frac{1-\theta}{\lambda}\right) u_e^2$$

The ISAIAH model was set up to solve for required jet power at start of trip, as a basis for estimating the EOTV mass.

$$\text{Trip time} = T_1 + T_2 = \frac{(\mu_a - 1) M_2 u_e^2}{2 p_{j1} \left(\frac{1-\theta}{\lambda}\right)} + \frac{(\mu_b - 1) M_3 u_e^2}{2 p_{j2} \left(\frac{1-\theta}{\lambda}\right)}$$

but $M_2 = \mu_b M_3$

$M_3 = M_{\text{OTV ARRIVE GEO}}$

and $P_{j2} = \Phi_3 P_{j1}$

APPENDIX A

So $$I = \frac{(\mu_a - 1)\mu_b u_e^2 M_3}{2 P_{j1}\left(\frac{1-\theta}{\lambda}\right)} + \frac{(\mu_b - 1) u_e^2 M_3}{2 \phi_3 P_{j1}\left(\frac{1-\theta}{\lambda}\right)}$$

$$= \frac{\lambda u_e^2}{2 P_{j1}(1-\theta)} \left[(\mu_a - 1)\mu_b + \frac{(\mu_b - 1)}{\phi_3} \right] M_{OTV}$$

and solving for P_{j1}, required jet power:

$$P_{j1} = \frac{\lambda u_e^2}{2 I (1-\theta)} \left[(\mu^{\tau} - 1)\mu^{(1-\tau)} + \frac{(\mu^{(1-\tau)} - 1)}{\Phi_3} \right] M_{OTV}$$

where τ is trip time split factor = 0.35

APPENDIX B

SELECTED EOTV DATA

PARAMETER VARIATIONS:
ELECTRIC SPECIFIC IMPULS VALUE = 8.[illegible]E+03
UP TRIP TIME VALUE = 1.8[illegible]E+02

Ion engine (reference) EOTV with 75-micron (3-mil) cell covers and solar array annealing. No thermal degradation or time delay.

SOLUTION RESULTS

No.	Parameter		Value		Equivalent
1	ONE-WAY MASS RATIO	=	1.[illegible]		
2	GROSS PAYLOAD RATIO	=	3.3[illegible]E+0[illegible]		
3	MASS RATIO FUNCTION	=	1.[illegible]E-01		
4	POWER FUNCTION	=	2.[illegible]E+0[illegible]		
5	REQUIRED JET POWER	=	1.[illegible]E+02	MEGAWATT	
6	RET. DEGRAD. CUM PWR RAT	=	[illegible].18[illegible]E-01		
7	UP DEGRAD. POWER RATIO	=	8.18[illegible]E-01		
8	RETURN TRIP FLUENCE	=	2.[illegible]E+16	E/CM2	
9	UP TRIP FLUENCE	=	1.[illegible]E+17	E/CM2	
10	RETURN LOG CUM FLUENCE	=	1.6[illegible]E+01		
11	UP TRIP LOG FLUENCE	=	1.71[illegible]E+01		
12	FLUENCE FOR 1ST WAY TRIP	=	1.[illegible]E+17	E/CM2	
13	ARRAY MASS/AREA	=	6.2[illegible]E-01	KG/M2	(1.2[illegible]E-01 LB/FT2)
14	TOTAL UP+RET. POWER RATI	=	[illegible]E-01		
15	THRUSTER-PPU EFFICIENCY	=	7.1[illegible]E-01		
16	LOCALIZED JET POWER	=	1.[illegible]E+02	MEGAWATT	
17	THRUSTER SPECIFIC MASS	=	[illegible].3[illegible]E-01	KG/KWE	(9.576E-01 LB/KWE)
18	ZETA CHARACTERISTIC	=	1.132[illegible]E-0[illegible]	KG/WATT	(2.[illegible]E-02 LB/WATT)
19	DUMMY	=	1.[illegible]E+0[illegible]		
20	POWER PROC SPECIFIC MASS	=	1.[illegible]E+0[illegible]	KG/KWE	(4.2[illegible]E+0[illegible] LB/KWE)
21	AUX SYS SPEC MASS	=	1.2[illegible]E-0[illegible]	KG/KWE	(2.753E-01 LB/KWE)
22	ARRAY SPEC MASS	=	3.7[illegible]E+0[illegible]	KG/KWE	(8.25[illegible]E+00 LB/KWE)
23	DSGN POWER RATIO/RATIO	=	1.[illegible]E+0[illegible]		
24	DESGN PRIOR LOG FLUENCE	=	1.[illegible]E+01		
25	EOTV FIXED MASS	=	1.155[illegible]E+03	TONNES	(2.[illegible]E+06 LB)
26	RETURN PAYLOAD	=	2.[illegible]E+02	TONNES	(4.4[illegible]9E+05 LB)
27	DESIGN JET POWER	=	1.[illegible]E+02	MEGAWATT	
28	AVG JET POWER UP	=	[illegible].62[illegible]E+01	MEGAWATT	
29	DESIGN ELEC POWER	=	1.44[illegible]E+02	MEGAWATT	
30	ARRAY DESIGN POWER	=	2.43[illegible]E+02	MEGAWATT	
31	RETURN TRIP TIME	=	4.1[illegible]E+01	DAYS	(9.99[illegible]E+02 HRS)
32	EOTV MASS GEO ARRIVE	=	5.333[illegible]E+03	TONNES	(1.176E+07 LB)
33	RETURN TRIP TIME TERM	=	2.55[illegible]E+0[illegible]		
34	UP PROPELLANT	=	4.57[illegible]E+02	TONNES	(1.[illegible]31E+06 LB)
35	RETURN PROPELLANT	=	1.23[illegible]E+02	TONNES	(2.725E+05 LB)
36	PROPELLANT SYSTEM MASS	=	5.6[illegible]E+01	TONNES	(1.245E+05 LB)
37	PWR GEN & DISTR MASS	=	7.7[illegible]E+02	TONNES	(1.7[illegible]E+06 LB)
38	PPL SYSTEM MASS	=	2.53[illegible]E+02	TONNES	(6.26[illegible]E+05 LB)
39	THRUSTER & INSTL MASS	=	7.6[illegible]E+01	TONNES	(1.677E+05 LB)
40	AUX SYSTEM MASS	=	1.45[illegible]E+01	TONNES	(4.[illegible]97E+04 LB)
41	EOTV EMPTY MASS	=	1.21[illegible]E+03	TONNES	(2.[illegible]67E+06 LB)
42	TOTAL PROPELLANT MASS	=	5.912E+02	TONNES	(1.313E+06 LB)
43	OTV STARTBURN MASS NO PL	=	1.[illegible]E+03	TONNES	(3.97[illegible]E+06 LB)
44	BLANKET MASS	=	4.394[illegible]E+02	TONNES	(1.011E+06 LB)
45	BLANKET STRUC	=	9.9[illegible]2[illegible]E+01	TONNES	(2.2[illegible]3E+05 LB)
46	BUSSING & MISC	=	3.4[illegible]E+01	TONNES	(7.685E+04 LB)
47	POWER PROCESSORS MASS	=	1.7[illegible]E+02	TONNES	(3.937E+05 LB)
48	PPL THERMAL CONTROL MASS	=	7.12[illegible]E+01	TONNES	(1.571E+05 LB)
49	INSTRUMENTATION MASS	=	1.21[illegible]E+01	TONNES	(2.667E+04 LB)
50	THRUSTERS MASS	=	6.46[illegible]E+01	TONNES	(1.425E+05 LB)
51	PROPELLANT SYS FRACTION	=	9.553E-02		
52	ARGON TANKS MASS	=	1.[illegible]65[illegible]E+01	TONNES	(2.349E+04 LB)
53	LO2/LH2 TANKS MASS	=	1.754E+0[illegible]	TONNES	(3.867E+03 LB)
54	FEED SYS MASS	=	1.192[illegible]E+01	TONNES	(2.[illegible]7E+04 LB)
55	LIQ & GAS RESID MASS	=	1.162[illegible]E+01	TONNES	(2.[illegible]7E+04 LB)
56	AUX PROP SYS MASS	=	1.47[illegible]E+01	TONNES	(3.258E+04 LB)
57	OPS STRUCTURE MASS	=	5.[illegible]97[illegible]E+01	TONNES	(1.124E+05 LB)
58	LO2/LH2 FRACTION	=	9.49[illegible]E-02		
59	LO2/LH2 PROP MASS	=	5.[illegible]E+01	TONNES	(1.289E+05 LB)
60	ARGON PROP MASS	=	5.32[illegible]E+02	TONNES	(1.174E+06 LB)
61	ARRAY AREA	=	1.37[illegible]E+06	M2	(1.474E+07 FT2)
62	THRUST PER THRUSTER	=	2.454E+0[illegible]	N	(5.416E-01 LBF)
63	THRUSTER INST CURRENT	=	7.777E+01	AMPS	
64	TOTAL THRUST	=	2.724E+03	N	(6.122E+02 LBF)
65	THRUST PER CORNER	=	6.8[illegible]E+02	N	(1.531E+02 LBF)
66	TOTAL NO. OF THRUSTERS	=	9.543E+0[illegible]		
67	NO. OF THRUS. PER/CORNER	=	2.386E+02		
68	SUPPLY VOLTAGE	=	1.61[illegible]E+03	VOLTS	
69	DUMMY	=	1.[illegible]E+0[illegible]		
70	THRUST INSTL SPEC MASS	=	5.11[illegible]E-01	KG/KWE	(1.127E+0[illegible] LB/KWE)
71	HLLV FLTS TO LIFT OTV	=	3.[illegible]E+0[illegible]		
72	HLLV FLTS TO REFUEL	=	1.55[illegible]E+0[illegible]		
73	OPS TOTAL MASS	=	4.35[illegible]E+02	TONNES	(9.591E+05 LB)
74	P GEN & D SYS COST	=	9.867[illegible]E+01	MILLION	
75	OPS COST	=	1.2[illegible]E+02	MILLION	
76	TRIP TIME COST	=	1.[illegible]74[illegible]E+01	MILLION	
77	HLLV COST TO LIFT OTV	=	3.72[illegible]E+01	MILLION	
78	HLLV COST TO REFUEL	=	1.819[illegible]E+02	MILLION	
79	HLLV COST TO LIFT PL	=	1.231E+02	MILLION	
80	EOTV CAP RECOV COST/FLT	=	3.411E+01	MILLION	
81	AMORTIZATION TIME PERIOD	=	6.8[illegible]E+0[illegible]	YEARS	
82	TOTAL ROUND TRIP TIME	=	2.517E+02	DAYS	(6.[illegible]4[illegible]E+03 HRS)
83	EOTV TOTAL CAP COST	=	2.894E+02	MILLION	
84	PAYLOAD COST	=	3.2[illegible]E+02	MILLION	
85	DIRECT COST/FLT	=	2.419E+01	MILLION	
86	EOTV TOTAL COST/FLT	=	[illegible].3[illegible]E+01	MILLION	
87	TOTAL TRANSP COST	=	2.[illegible]61E+02	MILLION	
88	TOTAL TRANSP COST/KG	=	5.725E+01	$/KG	(2.597E+01 $/LB)
89	AVERAGE/1ST TRIP TIME RA	=	1.15[illegible]E+0[illegible]		
90	ARRAY COST/AREA	=	7.2[illegible]E+01	DOLLARS	

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PARAMETER VARIATION:
ELECTRIC SPECIFIC IMPULS VALUE = 7.003E+03
UP TRIP TIME VALUE = 1.950E+02

Ion engine EOTV with 150-micron (6-mil) cell covers, no annealing. No thermal degradation or time delay.

SOLUTION RESULTS

1 ONE-WAY MASS RATIO	=	1.131E+00				
2 GROSS PAYLOAD RATIO	=	2.724E+00				
3 MASS RATIO FUNCTION	=	1.120E-01				
4 POWER FUNCTION	=	1.014E+02				
5 REQUIRED JET POWER	=	8.987E+01	MEGAWATT			
6 RET. DEGRAD. CUM PWR RAT	=	7.818E-01				
7 UP DEGRAD. POWER RATIO	=	8.436E-01				
8 RETURN TRIP FLUENCE	=	1.120E+16	E/CM2			
9 UP TRIP FLUENCE	=	4.221E+16	E/CM2			
10 RETURN LOG CUM FLUENCE	=	1.665E+01				
11 UP TRIP LOG FLUENCE	=	1.673E+01				
12 FLUENCE FOR 180-DAY TRIP	=	3.977E+16	E/CM2			
13 ARRAY MASS/AREA	=	1.060E+00	KG/M2	(	2.171E-01 LB/FT2	)
14 TOTAL UP+RET. POWER RATI	=	6.589E-01				
15 THRUSTER-PPU EFFICIENCY	=	6.956E-01				
16 IDEALIZED JET POWER	=	1.151E+02	MEGAWATT			
17 THRUSTER SPECIFIC MASS	=	5.702E-01	KG/KWE	(	1.257E+00 LB/KWE	)
18 ZETA CHARACTERISTIC	=	1.630E-02	KG/WATT	(	3.593E-02 LB/WATT	)
19 DUMMY	=	1.133E+00				
20 POWER PROC SPECIFIC MASS	=	1.912E+00	KG/KWE	(	4.216E+00 LB/KWE	)
21 AUX SYS SPEC MASS	=	1.628E-01	KG/KWE	(	3.590E-01 LB/KWE	)
22 ARRAY SPEC MASS	=	6.313E+00	KG/KWE	(	1.392E+01 LB/KWE	)
23 DEGR POWER RATIO/PRIOR	=	1.000E+00				
24 DEGR PRIOR LOG FLUENCE	=	1.400E+01				
25 IEOTV FIXED MASS	=	1.390E+03	TONNES	(	3.065E+06 LB	)
26 RETURN PAYLOAD	=	2.000E+02	TONNES	(	4.409E+05 LB	)
27 DESIGN JET POWER	=	8.987E+01	MEGAWATT			
28 AVG JET POWER UP	=	8.774E+01	MEGAWATT			
29 DESIGN ELEC POWER	=	1.292E+02	MEGAWATT			
30 ARRAY DESIGN POWER	=	1.946E+02	MEGAWATT			
31 RETURN TRIP TIME	=	5.161E+01	DAYS	(	1.239E+03 HRS	)
32 EOTV MASS GEO ARRIVE	=	5.638E+03	TONNES	(	1.243E+07 LB	)
33 RETURN TRIP TIME TERM	=	2.672E+06				
34 UP PROPELLANT	=	5.684E+02	TONNES	(	1.253E+06 LB	)
35 RETURN PROPELLANT	=	1.687E+02	TONNES	(	3.719E+05 LB	)
36 PROPELLANT SYSTEM MASS	=	7.035E+01	TONNES	(	1.551E+05 LB	)
37 PWR GEN & DISTR MASS	=	1.058E+03	TONNES	(	2.332E+06 LB	)
38 PPU SYSTEM MASS	=	2.465E+02	TONNES	(	5.435E+05 LB	)
39 THRUSTER & INSTL MASS	=	8.664E+01	TONNES	(	1.910E+05 LB	)
40 AUX SYSTEM MASS	=	2.098E+01	TONNES	(	4.625E+04 LB	)
41 IEOTV EMPTY MASS	=	1.469E+03	TONNES	(	3.238E+06 LB	)
42 TOTAL PROPELLANT MASS	=	7.365E+02	TONNES	(	1.624E+06 LB	)
43 OTV STARTBURN MASS NO PL	=	2.205E+03	TONNES	(	4.861E+06 LB	)
44 BLANKET MASS	=	8.625E+02	TONNES	(	1.901E+06 LB	)
45 BLANKET STRUC	=	1.347E+02	TONNES	(	2.970E+05 LB	)
46 BUSSING & MISC	=	4.699E+01	TONNES	(	1.036E+05 LB	)
47 POWER PROCESSORS MASS	=	1.551E+02	TONNES	(	3.419E+05 LB	)
48 PPU THERMAL CONTROL MASS	=	6.188E+01	TONNES	(	1.364E+05 LB	)
49 INSTRUMENTATION MASS	=	1.469E+01	TONNES	(	3.238E+04 LB	)
50 THRUSTERS MASS	=	7.364E+01	TONNES	(	1.624E+05 LB	)
51 PROPELLANT SYS FRACTION	=	9.553E-02				
52 ARGON TANKS MASS	=	1.327E+01	TONNES	(	2.926E+04 LB	)
53 LO2/LH2 TANKS MASS	=	2.185E+00	TONNES	(	4.817E+03 LB	)
54 FEED SYS MASS	=	1.473E+01	TONNES	(	3.247E+04 LB	)
55 LIQ & GAS RESID MASS	=	1.473E+01	TONNES	(	3.247E+04 LB	)
56 AUX PROP SYS MASS	=	1.841E+01	TONNES	(	4.059E+04 LB	)
57 EPS STRUCTURE MASS	=	4.949E+01	TONNES	(	1.091E+05 LB	)
58 LO2/LH2 FRACTION	=	9.898E-02				
59 LO2/LH2 PROP MASS	=	7.289E+01	TONNES	(	1.606E+05 LB	)
60 ARGON PROP MASS	=	6.636E+02	TONNES	(	1.463E+06 LB	)
61 ARRAY AREA	=	1.095E+06	M2	(	1.178E+07 FT2	)
62 THRUST PER THRUSTER	=	2.500E+00	N	(	5.620E-01 LBF	)
63 THRUSTER INST CURRENT	=	7.786E+01	AMPS			
64 TOTAL THRUST	=	2.618E+03	N	(	5.886E+02 LBF	)
65 THRUST PER CORNER	=	6.546E+02	N	(	1.472E+02 LBF	)
66 TOTAL NO. OF THRUSTERS	=	1.047E+03				
67 NO. OF THRUSTERS/CORNER	=	2.618E+02				
68 SUPPLY VOLTAGE	=	1.300E+03	VOLTS			
69 DUMMY	=	1.300E+00				
70 THRUST INSTL SPEC MASS	=	6.704E-01	KG/KWE	(	1.478E+00 LB/KWE	)
71 HLLV FLTS TO LIFT OTV	=	3.063E+00				
72 HLLV FLTS TO REFUEL	=	1.937E+00				
73 EPS TOTAL MASS	=	4.245E+02	TONNES	(	9.358E+05 LB	)
74 P GEN & D SYS COST	=	8.564E+01	MILLION			
75 EPS COST	=	1.256E+02	MILLION			
76 TRIP TIME COST	=	1.833E+01	MILLION			
77 HLLV COST TO LIFT OTV	=	4.519E+01	MILLION			
78 HLLV COST TO REFUEL	=	2.266E+01	MILLION			
79 HLLV COST TO LIFT PL	=	1.231E+02	MILLION			
80 EOTV CAP RECOV COST/FLT	=	3.793E+01	MILLION			
81 AMORTIZATION TIME PERIOD	=	7.573E+00	YEARS			
82 TOTAL ROUND TRIP TIME	=	2.766E+02	DAYS	(	6.639E+03 HRS	)
83 EOTV TOTAL CAP COST	=	2.814E+02	MILLION			
84 PAYLOAD COST	=	3.270E+02	MILLION			
85 DIRECT COST/FLT	=	3.206E+01	MILLION			
86 EOTV TOTAL COST/FLT	=	6.999E+01	MILLION			
87 TOTAL TRANSP COST	=	2.123E+02	MILLION			
88 TOTAL TRANSP COST/KG	=	5.888E+01	$/KG	(	2.671E+01 $/LB	)
89 AVERAGE/1ST TRIP TIME RA	=	1.303E+00				
90 ARRAY COST/AREA	=	7.925E+01	DOLLARS			

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PARAMETER VARIATION:
ELECTRIC SPECIFIC IMPULSE VALUE = 7.[illegible]E+03
UP TRIP TIME VALUE = 1.[illegible]E+02

SOLUTION RESULTS

Ion engine EOTV with 150-micron (6-mil) cell covers, no annealing. Includes solar array thermal degradation and startup delay.

1	ONE-WAY MASS RATIO	=	1.119E+0[illegible]				
2	GROSS PAYLOAD RATIO	=	1.98[illegible]				
3	MASS RATIO FUNCTION	=	1.5[illegible]E-01				
4	POWER FUNCTION	=	1.44[illegible]E-02				
5	REQUIRED JET POWER	=	1.71[illegible]E+02	MEGAWATT			
6	RET. DEGRAD. CUM PWR RAT	=	7.672[illegible]E-01				
7	UP DEGRAD. POWER RATIO	=	[illegible]				
8	RETURN TRIP FLUENCE	=	1.150E+14	E/CM2			
9	UP TRIP FLUENCE	=	4.221E+16	E/CM2			
10	RETURN LOG CUM FLUENCE	=	1.613E+01				
11	UP TRIP LOG FLUENCE	=	1.[illegible]E+01				
12	FLUENCE FOR 1ST-DAY TRIP	=	1.9[illegible]E+1[illegible]	E/CM2			
13	ARRAY MASS/AREA	=	1.[illegible]	KG/M2	(	[illegible] LB/FT2	)
14	TOTAL UPDEGR. POWER RATI	=	[illegible]				
15	THRUSTER-PPU EFFICIENCY	=	[illegible]				
16	IDEALIZED JET POWER	=	1.[illegible]E+02	MEGAWATT			
17	THRUSTER SPECIFIC MASS	=	5.7[illegible]E-01	KG/KWE	(	1.257E+00 LB/KWE	)
18	BETA CHARACTERISTIC	=	1.65[illegible]E-03	KG/WATT	(	3.65[illegible]E-03 LB/WATT	)
19	DUMMY	=	1.[illegible]				
20	POWER PROC SPECIFIC MASS	=	1.[illegible]	KG/KWE	(	4.2[illegible] LB/KWE	)
21	AUX SYS SPEC MASS	=	1.534E-01	KG/KWE	(	3.391E-01 LB/KWE	)
22	ARRAY SPEC MASS	=	6.311E+00	KG/KWE	(	1.392E+01 LB/KWE	)
23	DEGR POWER RATIO PRIOR	=	1.[illegible]				
24	DEGR PRIOR LOG FLUENCE	=	1.[illegible]E+01				
25	EOTV STAGE MASS	=	1.914E+03	TONNES	(	4.219E+06 LB	)
26	RETURN PAYLOAD	=	2.[illegible]E+02	TONNES	(	4.[illegible]E+05 LB	)
27	DESIGN JET POWER	=	1.21[illegible]E+02	MEGAWATT			
28	AVG JET POWER UP	=	1.1[illegible]E+02	MEGAWATT			
29	DESIGN ELEC POWER	=	1.765E+02	MEGAWATT			
30	ARRAY DESIGN POWER	=	2.67[illegible]E+02	MEGAWATT			
31	RETURN TRIP TIME	=	[illegible]E+01	DAYS	(	1.[illegible]E+03 HRS	)
32	EOTV MASS 6.0 ARRIVE	=	6.271E+03	TONNES	(	1.383E+07 LB	)
33	RETURN TRIP TIME TERM	=	2.[illegible]E+06				
34	UP PROPELLANT	=	7.346E+02	TONNES	(	1.620E+06 LB	)
35	RETURN PROPELLANT	=	2.6[illegible]E+02	TONNES	(	5.791E+05 LB	)
36	PROPELLANT SYSTEM MASS	=	9.73[illegible]E+01	TONNES	(	2.147E+05 LB	)
37	PWR GEN & DISTR MASS	=	1.[illegible]E+03	TONNES	(	2.[illegible]E+06 LB	)
38	PPU SYSTEM MASS	=	3.33[illegible]E+02	TONNES	(	7.345E+05 LB	)
39	THRUSTER & INSTL MASS	=	1.17[illegible]E+02	TONNES	(	2.581E+05 LB	)
40	AUX SYSTEM MASS	=	2.68[illegible]E+01	TONNES	(	5.921E+04 LB	)
41	EOTV EMPTY MASS	=	2.[illegible]E+03	TONNES	(	4.430E+06 LB	)
42	TOTAL PROPELLANT MASS	=	9.991E+02	TONNES	(	2.2[illegible]E+06 LB	)
43	OTV STARTBURN MASS NO PL	=	3.[illegible]E+03	TONNES	(	6.636E+06 LB	)
44	BLANKET MASS	=	1.187E+03	TONNES	(	2.616E+06 LB	)
45	BLANKET STRUC	=	1.853E+02	TONNES	(	4.086E+05 LB	)
46	BUSSING & MISC	=	6.465E+01	TONNES	(	1.425E+05 LB	)
47	POWER PROCESSORS MASS	=	2.[illegible]E+02	TONNES	(	4.62[illegible]E+05 LB	)
48	PPU THERMAL CONTROL MASS	=	8.362E+01	TONNES	(	1.844E+05 LB	)
49	INSTRUMENTATION MASS	=	2.[illegible]E+01	TONNES	(	4.430E+04 LB	)
50	THRUSTERS MASS	=	9.953E+01	TONNES	(	2.194E+05 LB	)
51	PROPELLANT SYS FRACTION	=	9.744E-02				
52	ARGON TANKS MASS	=	1.451E+01	TONNES	(	3.199E+04 LB	)
53	LO2/LH2 TANKS MASS	=	8.2[illegible]	TONNES	(	1.81[illegible]E+04 LB	)
54	FEED SYS MASS	=	1.994E+01	TONNES	(	4.4[illegible]E+04 LB	)
55	LIQ & GAS RESID MASS	=	1.999E+01	TONNES	(	4.4[illegible]E+04 LB	)
56	AUX PROP SYS MASS	=	2.45[illegible]E+01	TONNES	(	5.[illegible]E+04 LB	)
57	EPS STRUCTURAL MASS	=	6.7[illegible]E+01	TONNES	(	1.479E+05 LB	)
58	LO2/LH2 FRACTION	=	2.730E-01				
59	LO2/LH2 PROP MASS	=	2.735E+02	TONNES	(	6.032E+05 LB	)
60	ARGON PROP MASS	=	7.254E+02	TONNES	(	1.599E+06 LB	)
61	ARRAY AREA	=	1.5[illegible]E+06	M2	(	1.621E+07 FT2	)
62	THRUST PER THRUSTER	=	2.5[illegible]E+00	N	(	5.62[illegible]E-01 LBF	)
63	THRUSTER INST CURRENT	=	7.785E+01	AMPS			
64	TOTAL THRUST	=	3.539E+03	N	(	7.955E+02 LBF	)
65	THRUST PER CORNER	=	8.847E+02	N	(	1.989E+02 LBF	)
66	TOTAL NO. OF THRUSTERS	=	1.415E+03				
67	NO. OF THRUSTERS/CORNER	=	3.539E+02				
68	SUPPLY VOLTAGE	=	1.3[illegible]E+03	VOLTS			
69	DUMMY	=	1.[illegible]				
70	THRUST INSTL SPEC MASS	=	6.735E-01	KG/KWE	(	1.478E+00 LB/KWE	)
71	HLLV FLTS TO LIFT OTV	=	5.289E+01				
72	HLLV FLTS TO REFUEL	=	2.628E+01				
73	EPS TOTAL MASS	=	5.745E+02	TONNES	(	1.267E+06 LB	)
74	P GEN & D SYS COST	=	1.174E+02	MILLION			
75	EPS COST	=	1.63[illegible]E+02	MILLION			
76	TRIP TIME COST	=	1.065E+01	MILLION			
77	HLLV COST TO LIFT OTV	=	6.108E+01	MILLION			
78	HLLV COST TO REFUEL	=	3.074E+01	MILLION			
79	HLLV COST TO LIFT PL	=	1.231E+02	MILLION			
80	EOTV CAP RECOV COST/FLT	=	5.469E+01	MILLION			
81	AMORTIZATION TIME PERIOD	=	7.859E+00	YEARS			
82	TOTAL ROUND TRIP TIME	=	2.87[illegible]E+02	DAYS	(	6.88[illegible]E+03 HRS	)
83	EOTV TOTAL CAP COST	=	3.729E+02	MILLION			
84	PAYLOAD COST	=	3.20[illegible]E+02	MILLION			
85	DIRECT COST/FLT	=	4.[illegible]795E+01	MILLION			
86	EOTV TOTAL COST/FLT	=	1.[illegible]15E+02	MILLION			
87	TOTAL TRANSP COST	=	2.332E+02	MILLION			
88	TOTAL TRANSP COST/KG	=	6.476E+01	$/KG	(	2.938E+01 $/LB	)
89	AVERAGE/1ST TRIP TIME RA	=	1.3[illegible]E+0[illegible]				
90	ARRAY COST/AREA	=	7.825E+01	DOLLARS			

PARAMETER VARIATION:

SPECIFIC IMPULSE	VALUE =	5.250E+03
UP TRIP TIME	VALUE =	1.875E+02

MPD EOTV with 75-micron (3-mil) cell covers and solar array annealing. No thermal degradation or time delay.

SOLUTION RESULTS

No.	Item		Value	Units		Value	Units	
1	ONE-WAY MASS RATIO	=	1.130E+00					
2	GROSS PAYLOAD RATIO	=	2.954E+00					
3	MASS RATIO FUNCTION	=	1.554E-01					
4	POWER FUNCTION	=	5.272E-01					
5	REQUIRED JET POWER	=	7.150E+01	MEGAWATT				
6	RET. DEGRAD. CUM PWR RAT	=	7.111E-01					
7	UP DEGRAD. POWER RATIO	=	8.218E-01					
8	RETURN TRIP FLUENCE	=	2.744E+16	E/CM2				
9	UP TRIP FLUENCE	=	1.113E+17	E/CM2				
10	RETURN LOG CUM FLUENCE	=	1.644E+01					
11	UP TRIP LOG FLUENCE	=	1.714E+01					
12	FLUENCE FOR 180-DAY TRIP	=	1.[illegible]E+17	E/CM2				
13	ARRAY MASS/AREA	=	6.284E-01	KG/M2	(	1.287E-01	LB/FT2	)
14	TOTAL UP+RET. POWER RATI	=	5.044E-01					
15	THRUSTER-PPU EFFICIENCY	=	4.031E-01					
16	IDEALIZED JET POWER	=	1.[illegible]E+02	MEGAWATT				
17	THRUSTER SPECIFIC MASS	=	5.774E-01	KG/KWE	(	1.274E+00	LB/KWE	)
18	ZETA CHARACTERISTIC	=	1.892E-02	KG/WATT	(	4.172E-02	LB/WATT	)
19	STARTBURN LAMBDA*	=	1.530E-01					
20	POWER PROC SPECIFIC MASS	=	1.930E+00	KG/KWE	(	4.256E+00	LB/KWE	)
21	AUX SYS SPEC MASS	=	1.351E-01	KG/KWE	(	2.979E-01	LB/KWE	)
22	ARRAY SPEC MASS	=	3.742E+00	KG/KWE	(	8.250E+00	LB/KWE	)
23	DEGR POWER RATIO/PRIOR	=	1.39[illegible]E+00					
24	DEGR PRIOR LOG FLUENCE	=	1.4[illegible]E+01					
25	EOTV FIXED MASS	=	1.183E+03	TONNES	(	2.608E+06	LB	)
26	RETURN PAYLOAD	=	2.[illegible]E+02	TONNES	(	4.409E+05	LB	)
27	DESIGN JET POWER	=	7.154E+01	MEGAWATT				
28	AVG JET POWER UP	=	6.324E+01	MEGAWATT				
29	DESIGN ELEC POWER	=	1.482E+02	MEGAWATT				
30	ARRAY DESIGN POWER	=	2.451E+02	MEGAWATT				
31	RETURN TRIP TIME	=	4.631E+01	DAYS	(	1.1[illegible]E+03	HRS	)
32	EOTV MASS GEO ARRIVE	=	5.566E+03	TONNES	(	1.227E+07	LB	)
33	RETURN TRIP TIME TERM	=	2.554E+06					
34	UP PROPELLANT	-	7.659E+02	TONNES	(	1.677E+06	LB	)
35	RETURN PROPELLANT		2.123E+02	TONNES	(	4.680E+05	LB	)
36	PROPELLANT SYSTEM MASS	=	1.7[illegible]E+02	TONNES	(	3.766E+05	LB	)
37	PWR GEN & DISTR MASS	=	7.796E+02	TONNES	(	1.719E+06	LB	)
38	PPU SYSTEM MASS	=	2.827E+02	TONNES	(	6.232E+05	LB	)
39	THRUSTER & INSTL MASS	=	1.007E+02	TONNES	(	2.221E+05	LB	)
40	AUX SYSTEM MASS	=	2.002E+01	TONNES	(	4.414E+04	LB	)
41	EOTV EMPTY MASS	=	1.354E+03	TONNES	(	2.985E+06	LB	)
42	TOTAL PROPELLANT MASS	=	9.728E+02	TONNES	(	2.145E+06	LB	)
43	OTV STARTBURN MASS	=	6.327E+03	TONNES	(	1.395E+07	LB	)
44	BLANKET MASS	=	6.44[illegible]E+02	TONNES	(	1.420E+06	LB	)
45	BLANKET STRUC	=	1.006E+02	TONNES	(	2.217E+05	LB	)
46	BUSSING & MISC	=	3.508E+01	TONNES	(	7.735E+04	LB	)
47	POWER PROCESSORS MASS	=	1.778E+02	TONNES	(	3.920E+05	LB	)
48	PPU THERMAL CONTROL MASS	=	7.096E+01	TONNES	(	1.564E+05	LB	)
49	INSTRUMENTATION MASS	=	1.354E+01	TONNES	(	2.985E+04	LB	)
50	THRUSTERS MASS	=	8.56[illegible]E+01	TONNES	(	1.887E+05	LB	)
51	PROPELLANT SYS FRACTION	=	1.756E-01					
52	HYDROGEN TANKS MASS	=	8.766E+01	TONNES	(	1.933E+05	LB	)
53	LO2/LH2 TANKS MASS	=	2.886E+00	TONNES	(	6.363E+03	LB	)
54	FEED SYS MASS	=	1.946E+01	TONNES	(	4.289E+04	LB	)
55	LIQ & GAS RESID MASS	=	1.946E+01	TONNES	(	4.289E+04	LB	)
56	AUX PROP SYS MASS	=	2.432E+01	TONNES	(	5.362E+04	LB	)
57	EPS STRUCTURE MASS	=	6.596E+01	TONNES	(	1.454E+05	LB	)
58	LO2/LH2 FRACTION	=	9.890E-02					
59	LO2/LH2 PROP MASS	=	9.621E+01	TONNES	(	2.121E+05	LB	)
60	HYDROGEN PROP MASS	=	8.766E+02	TONNES	(	1.933E+06	LB	)
61	ARRAY AREA	=	1.379E+06	M2	(	1.484E+07	FT2	)
62	THRUST PER THRUSTER	=	5.819E+00	N	(	1.308E+00	LBF	)
63	THRUSTER INST CURRENT	=	3.399E+00	AMPS				
64	TOTAL THRUST	=	2.782E+03	N	(	6.255E+02	LBF	)
65	THRUST PER CORNER	=	6.951E+02	N	(	1.563E+02	LBF	)
66	TOTAL NO. OF THRUSTERS	=	4.778E+02					
67	NO. OF THRUSTERS/CORNER	=	1.195E+02					
68	SUPPLY VOLTAGE	=	4.516E+02	VOLTS				
69	DUMMY	=	1.000E+00					
70	THRUST INSTL SPEC MASS	=	6.797E-01	KG/KWE	(	1.499E+00	LB/KWE	)
71	HLLV FLTS TO LIFT OTV	=	3.561E+0[illegible]					
72	HLLV FLTS TO REFUEL	=	2.558E+0[illegible]					
73	EPS TOTAL MASS	=	5.743E+02	TONNES	(	1.266E+06	LB	)
74	P GEN & D SYS COST	=	9.931E+01	MILLION				
75	EPS COST	=	6.719E+01	MILLION				
76	TRIP TIME COST	=	1.789E+01	MILLION				
77	HLLV COST TO LIFT OTV	=	4.166E+01	MILLION				
78	HLLV COST TO REFUEL	=	2.993E+01	MILLION				
79	HLLV COST TO LIFT PL	=	1.231E+02	MILLION				
80	EOTV CAP RECOV COST/FLT	=	3.170E+01	MILLION				
81	AMORTIZATION TIME PERIOD	=	7.214E+00	YEARS				
82	TOTAL ROUND TRIP TIME	=	2.635E+02	DAYS	(	6.324E+03	HRS	)
83	EOTV TOTAL CAP COST	=	2.382E+02	MILLION				
84	PAYLOAD COST	=	3.200E+02	MILLION				
85	DIRECT COST/FLT	=	3.993E+01	MILLION				
86	EOTV TOTAL COST/FLT	=	8.952E+01	MILLION				
87	TOTAL TRANSP COST	=	2.126E+02	MILLION				
88	TOTAL TRANSP COST/KG	=	5.905E+01	$/KG	(	2.679E+01	$/LB	)
89	AVERAGE/1ST TRIP TIME RA	=	1.15[illegible]E+00					
9	ARRAY COST/AREA	=	7.204E+01	DOLLARS				

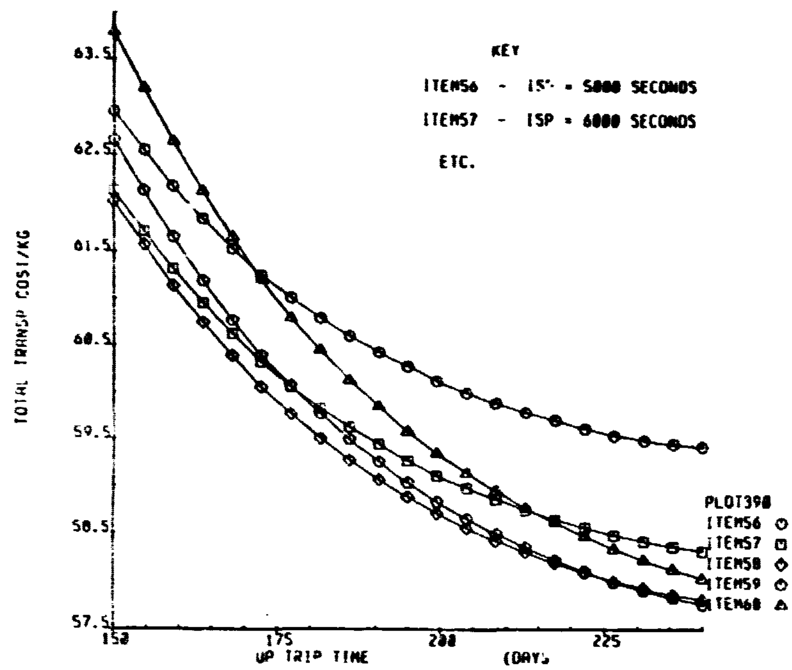
PARAMETER VARIATION:

SPECIFIC IMPULSE	VALUE =	5.25[illegible]E+03	
UP TRIP TIME	VALUE =	1.875E+02	

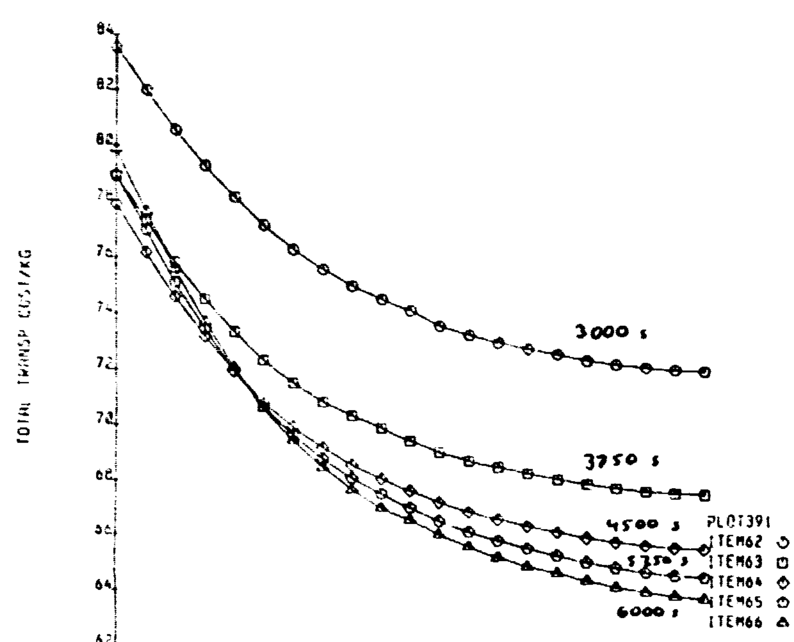
MPD EOTV with 150-micron (6-mil) cell covers, no annealing, startup delay, and solar array thermal degradation (EOTVM)

SOLUTION RESULTS

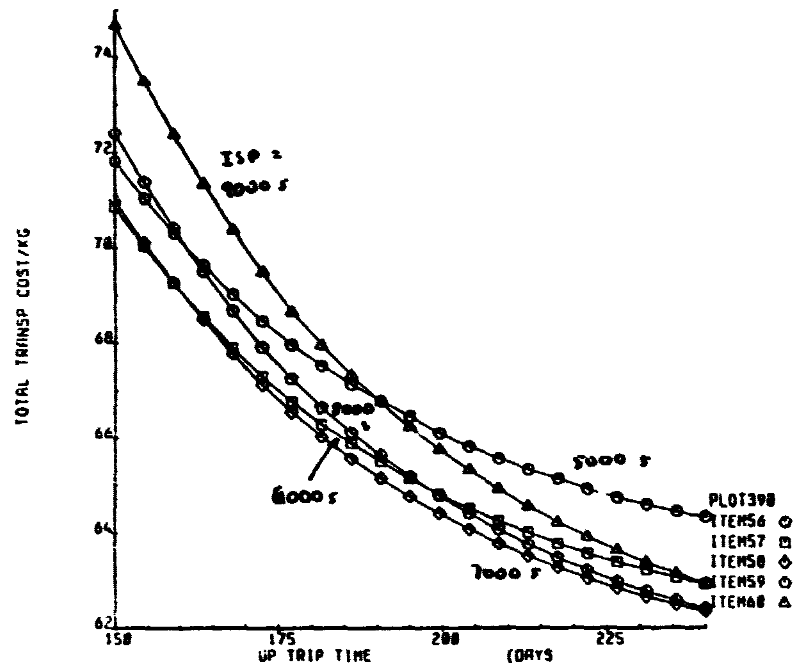
No.	Parameter		Value	Unit		English value	Unit	
1	ONE-WAY MASS RATIO	=	1.16[illegible]E+00					
2	GROSS PAYLOAD RATIO	=	1.416E+00					
3	MASS RATIO FUNCTION	=	1.730E-01					
4	POWER FUNCTION	=	8.679E+01					
5	REQUIRED JET POWER	=	1.115E+02	MEGAWATT				
6	RET. DEGRAD. CUM PWR RAT	=	7.572E-01					
7	UP DEGRAD. POWER RATIO	=	8.655E-01					
8	RETURN TRIP FLUENCE	=	1.538E+16	E/CM2				
9	UP TRIP FLUENCE	=	4.[illegible]58E+16	E/CM2				
10	RETURN LOG CUM FLUENCE	=	1.619E+01					
11	UP TRIP LOG FLUENCE	=	1.675E+01					
12	FLUENCE FOR 180-DAY TRIP	=	5.9[illegible]E+16	E/CM2				
13	ARRAY MASS/AREA	=	1.06[illegible]E+00	KG/M2	(	2.171E-01	LB/FT2	)
14	TOTAL UP+RET. POWER RATI	=	6.550E-01					
15	THRUSTER-PPU EFFICIENCY	=	4.831E-01					
16	IDEALIZED JET POWER	=	1.473E+02	MEGAWATT				
17	THRUSTER SPECIFIC MASS	=	5.779E-01	KG/KWE	(	1.274E+00	LB/KWE	)
18	ZETA CHARACTERISTIC	=	2.533E-02	KG/WATT	(	5.585E-02	LB/WATT	)
19	STARTBURN LAMBDA*	=	1.951E-01					
20	POWER PROC SPECIFIC MASS	=	1.908E+00	KG/KWE	(	4.206E+00	LB/KWE	)
21	AUX SYS SPEC MASS	=	1.54[illegible]E-01	KG/KWE	(	3.396E-01	LB/KWE	)
22	ARRAY SPEC MASS	=	6.315E+00	KG/KWE	(	1.392E+01	LB/KWE	)
23	DEGR POWER RATIO/PRIOR	=	1.00[illegible]E+00					
24	DEGR PRIOR LOG FLUENCE	=	1.48[illegible]E+01					
25	IEOTV FIXED MASS	=	2.557E+03	TONNES	(	5.637E+06	LB	)
26	RETURN PAYLOAD	=	2.00[illegible]E+02	TONNES	(	4.4[illegible]9E+05	LB	)
27	DESIGN JET POWER	=	1.115E+02	MEGAWATT				
28	AVG JET POWER UP	=	1.018E+02	MEGAWATT				
29	DESIGN ELEC POWER	=	2.3[illegible]8E+02	MEGAWATT				
30	ARRAY DESIGN POWER	=	3.586E+02	MEGAWATT				
31	RETURN TRIP TIME	=	7.063E+01	DAYS	(	1.695E+03	HRS	)
32	EOTV MASS GEO ARRIVE	=	7.309E+03	TONNES	(	1.611E+07	LB	)
33	RETURN TRIP TIME TERM	=	2.018E+06					
34	UP PROPELLANT	=	1.17[illegible]E+03	TONNES	(	2.579E+06	LB	)
35	RETURN PROPELLANT	=	4.841E+02	TONNES	(	1.067E+06	LB	)
36	PROPELLANT SYSTEM MASS	=	2.68[illegible]E+02	TONNES	(	5.907E+05	LB	)
37	PWR GEN & DISTR MASS	=	1.924E+03	TONNES	(	4.242E+06	LB	)
38	PPU SYSTEM MASS	=	4.404E+02	TONNES	(	9.709E+05	LB	)
39	THRUSTER & INSTL MASS	=	1.569E+02	TONNES	(	3.459E+05	LB	)
40	AUX SYSTEM MASS	=	3.556E+01	TONNES	(	7.840E+04	LB	)
41	IEOTV EMPTY MASS	=	2.825E+03	TONNES	(	6.228E+06	LB	)
42	TOTAL PROPELLANT MASS	=	1.654E+03	TONNES	(	3.646E+06	LB	)
43	OTV STARTBURN MASS	=	8.479E+03	TONNES	(	1.869E+07	LB	)
44	BLANKET MASS	=	1.589E+03	TONNES	(	3.504E+06	LB	)
45	BLANKET STRUC	=	2.482E+02	TONNES	(	5.472E+05	LB	)
46	MUSSING & MISC	=	8.659E+01	TONNES	(	1.9[illegible]9E+05	LB	)
47	POWER PROCESSORS MASS	=	2.772E+02	TONNES	(	6.107E+05	LB	)
48	PPU THERMAL CONTROL MASS	=	1.105E+02	TONNES	(	2.437E+05	LB	)
49	INSTRUMENTATION MASS	=	2.825E+01	TONNES	(	6.228E+04	LB	)
50	THRUSTERS MASS	=	1.334E+02	TONNES	(	2.940E+05	LB	)
51	PROPELLANT SYS FRACTION	=	1.620E-01					
52	HYDROGEN TANKS MASS	=	1.201E+02	TONNES	(	2.648E+05	LB	)
53	LO2/LH2 TANKS MASS	=	1.359E+01	TONNES	(	2.996E+04	LB	)
54	FEED SYS MASS	=	3.308E+01	TONNES	(	7.292E+04	LB	)
55	LIQ & GAS RESID MASS	=	3.308E+01	TONNES	(	7.292E+04	LB	)
56	AUX PROP SYS MASS	=	4.135E+01	TONNES	(	9.115E+04	LB	)
57	EPS STRUCTURE MASS	=	1.029E+02	TONNES	(	2.269E+05	LB	)
58	LO2/LH2 FRACTION	=	2.739E-01					
59	LO2/LH2 PROP MASS	=	4.53[illegible]E+02	TONNES	(	9.986E+05	LB	)
60	HYDROGEN PROP MASS	=	1.201E+03	TONNES	(	2.648E+06	LB	)
61	ARRAY AREA	=	2.017E+06	M2	(	2.171E+07	FT2	)
62	THRUST PER THRUSTER	=	5.819E+00	N	(	1.308E+00	LBF	)
63	THRUSTER INST CURRENT	=	3.899E+06	AMPS				
64	TOTAL THRUST	=	4.332E+03	N	(	9.737E+02	LBF	)
65	THRUST PER CORNER	=	1.083E+03	N	(	2.434E+02	LBF	)
66	TOTAL NO. OF THRUSTERS	=	7.444E+02					
67	NO. OF THRUSTERS/CORNER	=	1.861E+02					
68	SUPPLY VOLTAGE	=	8.516E+02	VOLTS				
69	DUMMY	=	1.02[illegible]E+07					
70	THRUST INSTL SPEC MASS	=	6.797E-01	KG/KWE	(	1.499E+00	LB/KWE	)
71	HLLV FLTS TO LIFT OTV	=	7.436E+00					
72	HLLV FLTS TO REFUEL	=	4.358E+00					
73	EPS TOTAL MASS	=	9.008E+02	TONNES	(	1.986E+06	LB	)
74	P GEN & D SYS COST	=	1.578E+02	MILLION				
75	EPS COST	=	1.054E+02	MILLION				
76	TRIP TIME COST	=	1.060E+01	MILLION				
77	HLLV COST TO LIFT OTV	=	8.693E+01	MILLION				
78	HLLV COST TO REFUEL	=	5.089E+01	MILLION				
79	HLLV COST TO LIFT PL	=	1.231E+02	MILLION				
80	EOTV CAP RECOV COST/FLT	=	5.173E+01	MILLION				
81	AMORTIZATION TIME PERIOD	=	7.006E+00	YEARS				
82	TOTAL ROUND TRIP TIME	=	2.880E+02	DAYS	(	6.913E+03	HRS	)
83	EOTV TOTAL CAP COST	=	3.001E+02	MILLION				
84	PAYLOAD COST	=	3.203E+02	MILLION				
85	DIRECT COST/FLT	=	6.089E+01	MILLION				
86	EOTV TOTAL COST/FLT	=	1.313E+02	MILLION				
87	TOTAL TRANSP COST	=	2.544E+02	MILLION				
88	TOTAL TRANSP COST/KG	=	7.066E+01	$/KG	(	3.205E+01	$/LB	)
89	AVERAGE/1ST TRIP TIME RA	=	1.300E+00					
90	ARRAY COST/AREA	=	7.025E+01	DOLLARS				



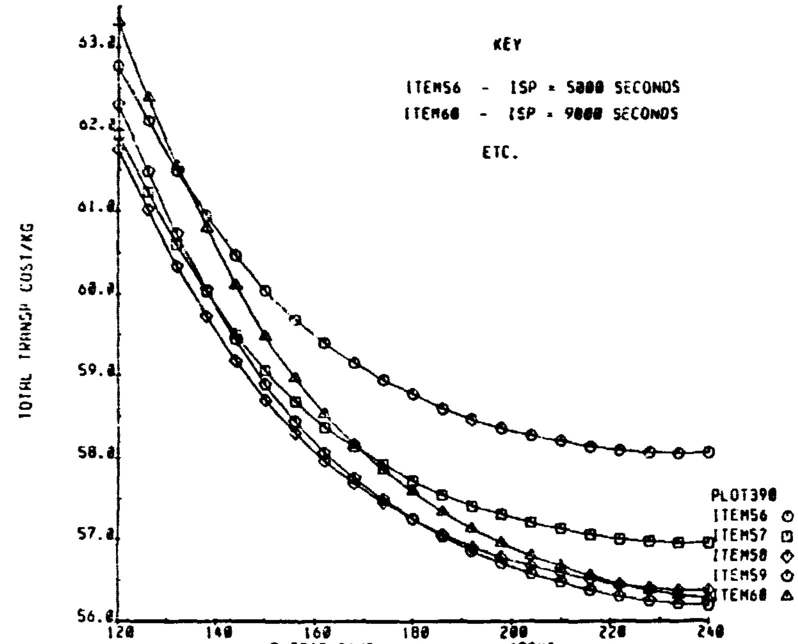
Ion engine EOTV with 150-micron (6-mil) cell covers, no annealing.
No thermal degradation or time delay.



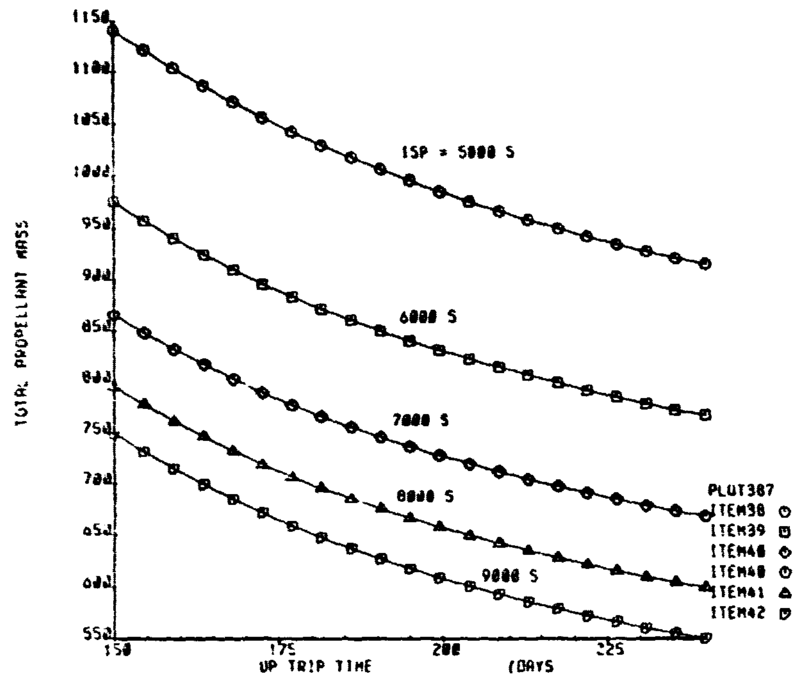
MPD EOTV with 150-micron (6-mil) cell covers, no annealing.
Includes solar array thermal degradation and time delay.



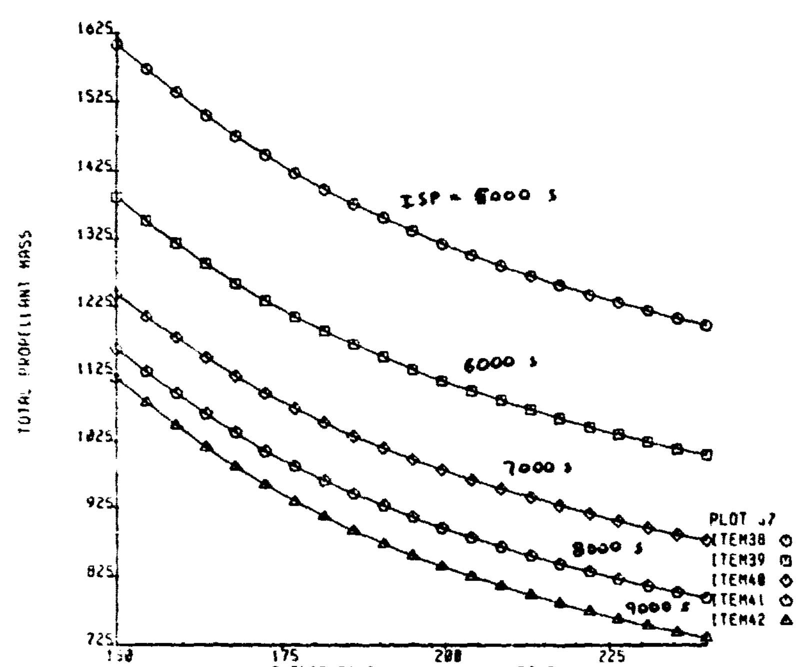
Ion engine EOTV with 150-micron (6-mil) cell covers, no annealing. Includes solar array thermal degradation and startup delay.



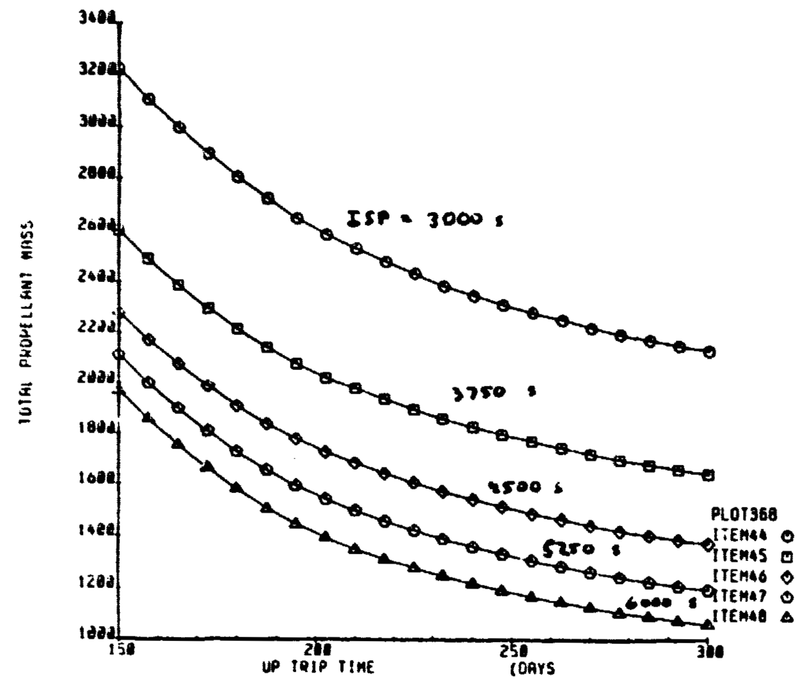
Ion engine (reference) EOTV with 75-micron (3-mil) cell covers and solar array annealing. No thermal degradation or time delay.



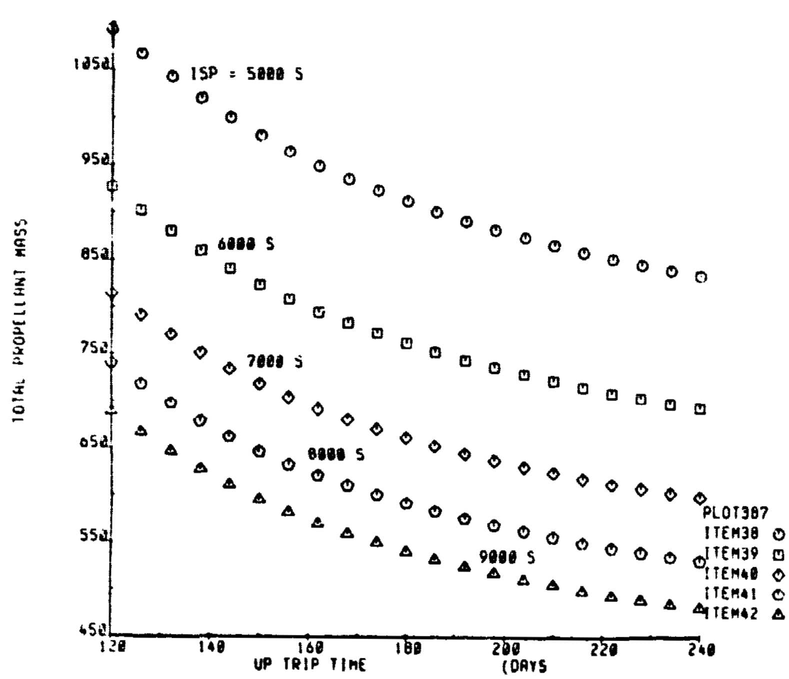
Ion engine EOTV with 150-micron (6-mil) cell covers, no annealing.
No thermal degradation or time delay.



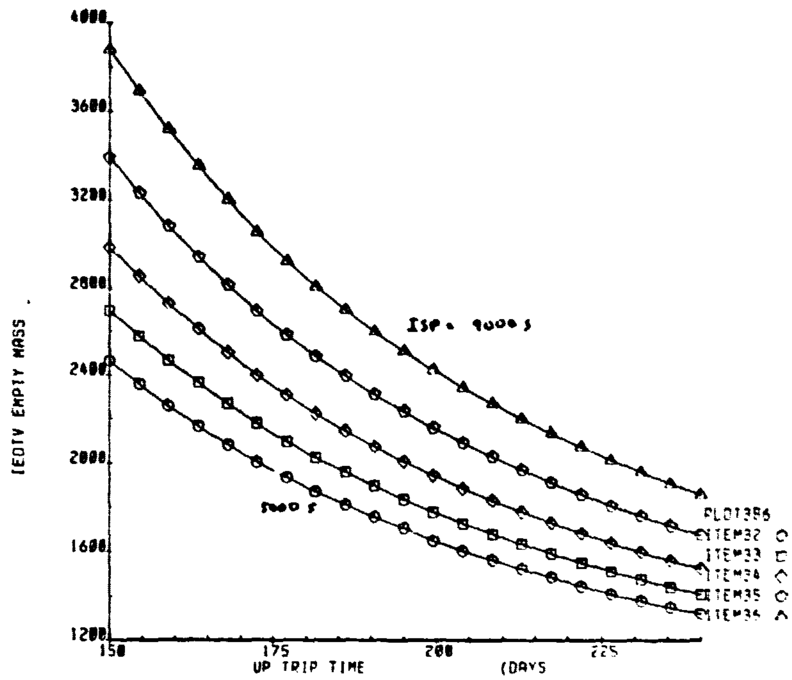
Ion engine EOTV with 150-micron (6-mil) cell covers, no annealing.
Includes solar array thermal degradation and startup delay.



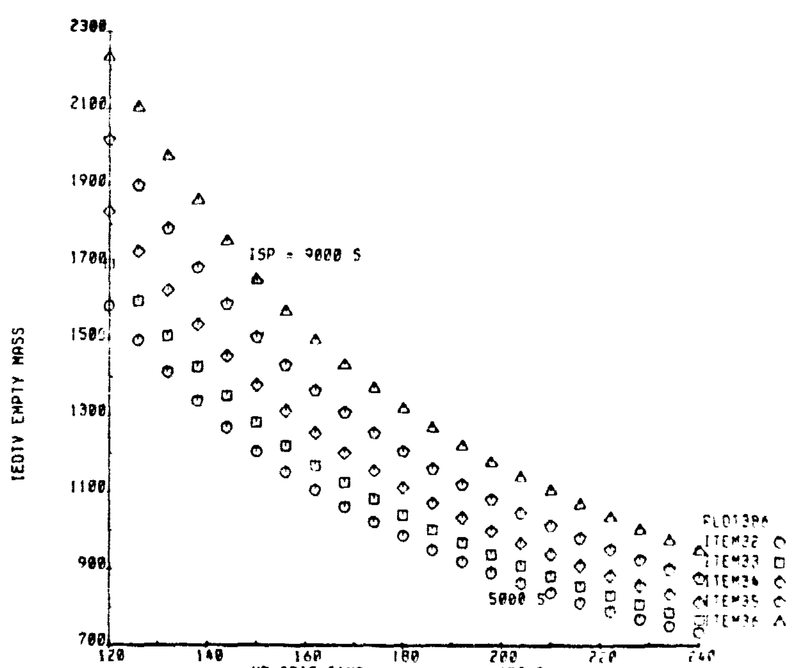
MPD EOTV with 150-micron (6-mil) cell covers, no annealing. Includes solar array thermal degradation and time delay.



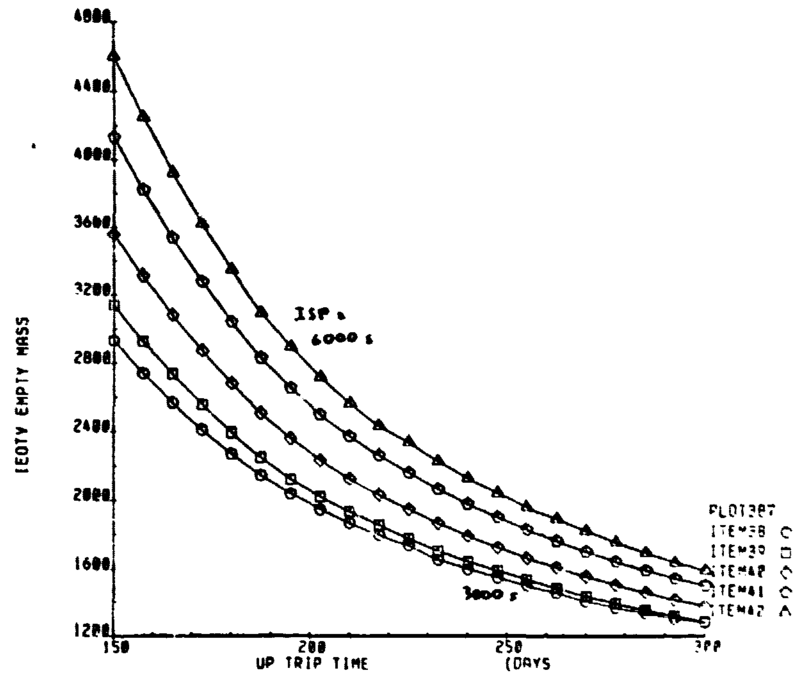
Ion engine (reference) EOTV with 75-micron (3-mil) cell covers and solar array annealing. No thermal degradation or time delay.



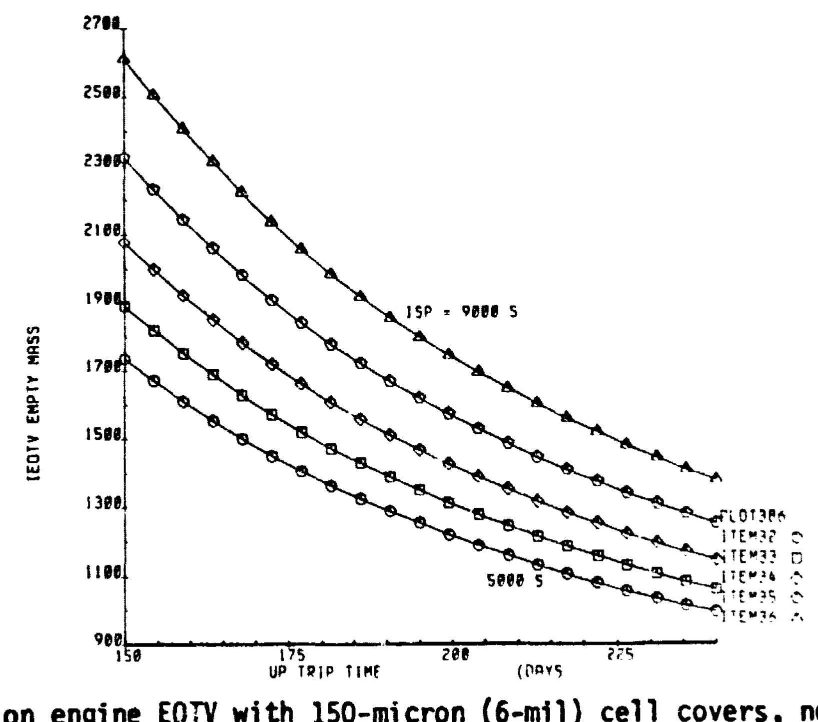
Ion engine EOTV with 150-micron (6-mil) cell covers, no annealing. Includes solar array thermal degradation and startup delay.



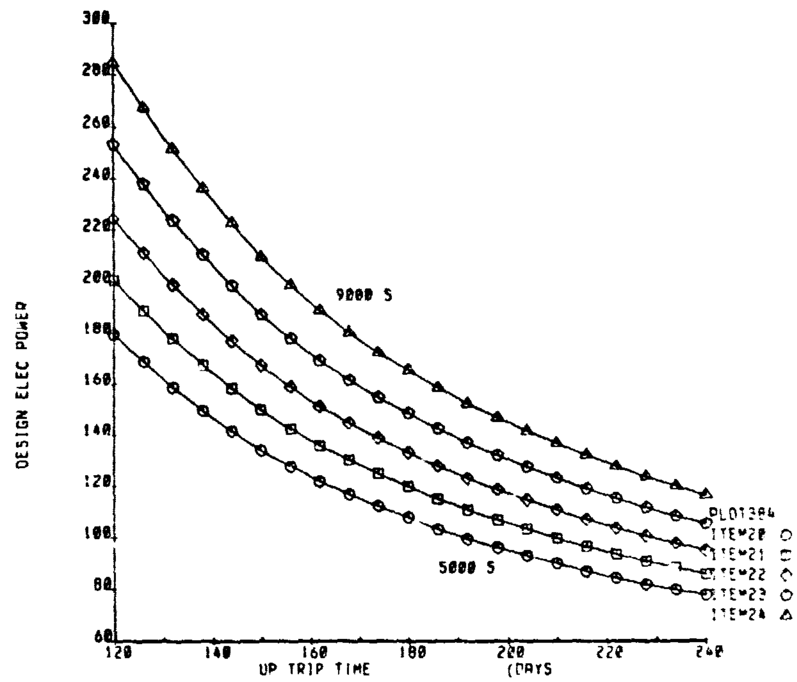
Ion engine (reference) EOTV with 75-micron (3-mil) cell covers and solar array annealing. No thermal degradation or time delay.



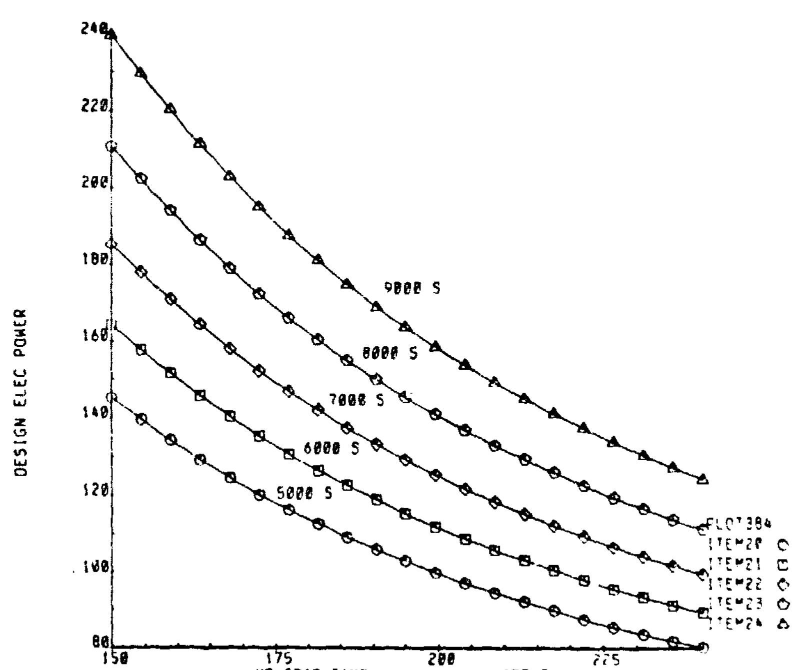
MPD EOTV with 150-micron (6-mil) cell covers, no annealing.
Includes solar array thermal degradation and time delay.



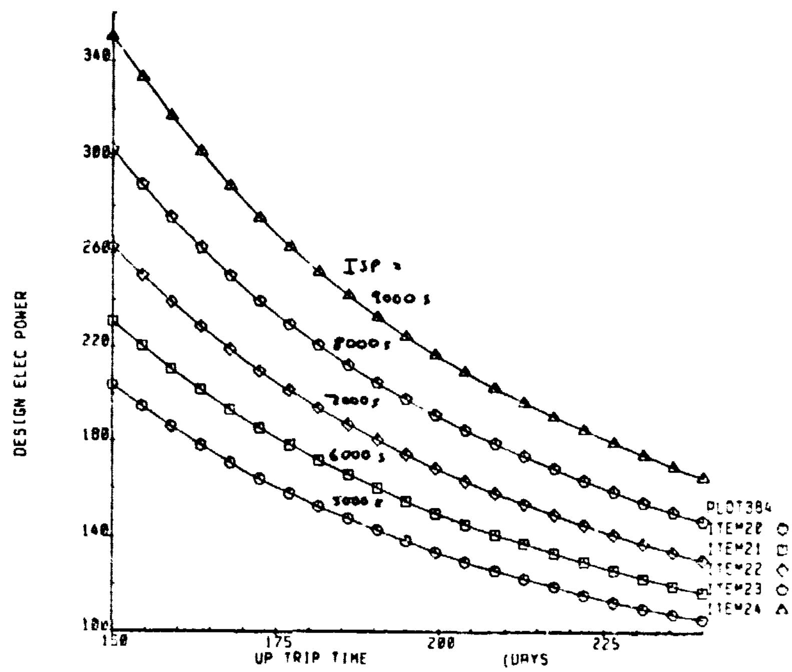
Ion engine EOTV with 150-micron (6-mil) cell covers, no annealing.
No thermal degradation or time delay.



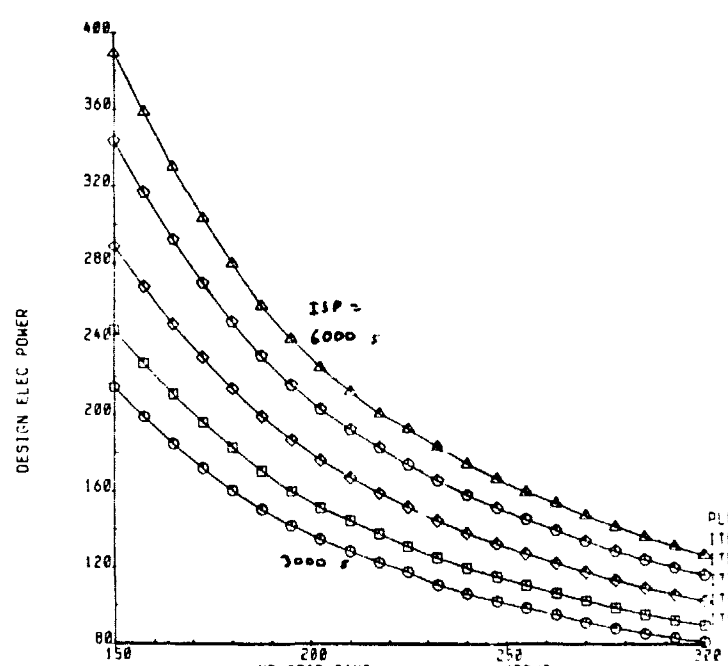
Ion engine EOTV with 150-micron (6-mil) cell covers, no annealing. No thermal degradation or time delay.



Ion engine (reference) EOTV with 75-micron (3-mil) cell covers and solar array annealing. No thermal degradation or time delay.



Ion engine EOTV with 150-micron (6-mil) cell covers, no annealing.
Includes solar array thermal degradation and startup delay.



MPD EOTV with 150-micron (6-mil) cell covers, no annealing.
Includes solar array thermal degradation and time delay.